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science Applications, INC.

(NASA-CR-155055) ADVANCED PLANETARY STUDIES N77-34072
Annul Report (Science Applications, Inc.)
87 p HC A05/MF A01 CSCL 03B
Unclas
G3/91 15083



Report No. SAI 1-120-399-A3

ADVANCED PLANETARY STUDIES
THIRD ANNUAL REPORT

by

Science Applications, Inc.
5005 Newport Drive, Suite 305
Rolling Meadows, Illinois 60008

for

Planetary Programs Division
Office of Space Science
NASA Headquarters
Washington, D.C.

Contract No. NASW-2783

March 1976

FOREWORD

This report summarizes the results of planetary advanced studies and planning support performed by Science Applications, Inc. (SAI) under Contract No. NASW-2783 for the Planetary Programs Division, Code SL, of NASA Headquarters during the twelve month period 1 February 1975 through 31 January 1976. A total effort of 8820 man-hours (54.4 man-months) was expended on five specific study tasks and one general support task. The total contract value was \$233,670, with 85% of the work performed by the staff of the SAI Chicago office. Inquiries regarding further information on the results reported here may be directed to the project leader, Mr. John Niehoff, at 312/253-5500.

Science Applications, Inc.
Schaumburg, Illinois

Advanced Planetary Studies
Third Annual Report

STAR Abstract

Results of planetary advanced studies and planning support provided by Science Applications, Inc. staff members to the Lunar and Planetary Programs Division of OSS/NASA for the period 1 February 1975 through 31 January 1976 are summarized. The scope of analyses includes cost estimation research, planetary mission performance, penetrator mission concepts for airless planets/satellites, geology orbiter payload adaptability, lunar mission performance, and advanced planning activities. This work covers 4.5 man-years of research. Study reports and related publications are included in a bibliography section.

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1. INTRODUCTION

Science Applications, Inc. (SAI) participates in a program of advanced concepts studies and planning analysis for the Planetary Programs Office, Code SL, of NASA Headquarters. SAI's charter is to provide unbiased preliminary analyses and evaluations for Code SL planning activities. Specifically, the objective of this support is to ensure that NASA has an adequate range of viable future planetary mission options in order to pursue the objectives of solar system exploration within the changing constraints of our space program. The nature of the work involved is quite varied, ranging from short quick response items to pre-Phase A level mission studies. During the past contract year a total of eight SAI staff members contributed to this effort.

The purpose of this report is to summarize the significant results generated under this advanced studies contract during the twelve month period, 1 February 1975 through 31 January 1976. Progress reports of the task efforts have been given at scheduled quarterly reviews, and in Code SL's Quarterly Newsletter. Task reports have been prepared and presentations given to a wide audience at NASA Headquarters, NASA Centers, and at technical meetings on the significant study results. This report, therefore, is necessarily brief, with the intention of directing previously uninformed interested readers to detailed documentation, and to serve as a future reference to previously completed advanced studies.

The next section of the report presents the individual tasks performed during the contract period and briefly describes each task presenting the key results and conclusions that were generated. The last section of the report is a bibliography of the reports and publications that have resulted from the task analyses. SAI is presently beginning another twelve month period of advanced studies for the Planetary Program Division with a schedule of six study tasks, several of which are continuing research on the work reported here.

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2. TASK SUMMARIES

An initial schedule of seven study tasks was planned for the twelve month contract period, 1 February 1975 through 31 January 1976. One of these tasks, Titan Mission Concepts Study, was a continuation of the previous year's work, but all further work was cancelled by agreement early in the contract period due to similar analyses being performed elsewhere for NASA. The remaining six tasks are listed below.

- 1) Advanced Planning Activities
- 2) Cost Estimation Research
- 3) Planetary Missions Performance Handbook--Volume II, Inner Planets
- 4) Penetrator Mission Concepts for Mercury and the Galilean Satellites
- 5) Geology Orbiter Comparison Study
- 6) Shuttle-Based Lunar Missions Performance Analysis

Task 1, Advanced Planning Activities, is a general support task designed to provide a budgeted level of effort for technical assistance on short term planning problems which frequently confront the Planetary Programs Division. The remaining five tasks are planned efforts with specific objectives of analysis.

A total of 8820 man-hours of effort (54.4 man-months) was expended in completing this schedule of tasks. A summary description and discussion of key results for each task is presented in the subsections which follow. The level of effort devoted to each task is given with the task title at the beginning of each subsection. Specific reports generated as part of this contract are noted in the list of publications to be found in Section 3 of this report.

2.1 Advanced Planning Activity (2462 Man-Hours)

The purpose of this task is to provide technical assistance to the Planetary Program Office on unscheduled planning activities which arise during the contract period. This type of advanced planning support is a traditional segment of the broader advanced studies work the staff at SAI have performed for Code SL during all past contract periods. The sub-tasks within this activity range from straightforward exchanges of technical data by phone, through several page responses by mail or telecopier, to more extensive memoranda and presentations, and occasionally to complete status reports on subjects of particular interest. The level of effort per sub-task can vary from as little as one man-hour to as much as three man-months. A total of 29 of the more significant advanced planning subtasks, performed during the recently completed contract period, are summarized here. Each of these was the subject of a written submission at the time of its completion. Descriptive titles of these subtasks are tabulated in chronological order in Table 1. A brief summary of each of these subtasks is presented in the sub-sections which follow.

2.1.1 Mars Penetrator Design Status Report

A brief oral report was prepared on the Mars penetrator design as an orientation presentation for the Committee on Planetary Exploration (COMPLEX) of the Space Science Board. It followed a summary to COMPLEX on the status of penetrator science instrumentation given by Dr. H. Mark, Director of Ames Research Center/NASA. As a topic of introduction, the characteristics of surface penetrability and their engineering design implications were discussed. Description of the Mars penetrator concept included a summary of the defining subsystem parameters, the total penetrator mass breakdown, power and communication subsystem details, and review of the various environments experienced by penetrator science instrumentation. Various power and communications trade-offs were discussed. Finally, the mass and guidance requirements of adopting the Mars penetrator design to a Mercury mission were introduced.

2.1.2 1975 Planetary Mission Model Performance Requirements

The purpose of this subtask was to prepare launch vehicle performance requirements for the new planetary mission model. In particular, injected mass and vis viva energy (C3) were required for each mission of the model for the period 1980-88. A total of 16 missions were examined. Mission transfer time was also determined from mission transfer data. Spacecraft retro performance calculations, assuming earth-storable propellants, were required when post-injection maneuvers were included in the mission concept. This was true for rendezvous, orbiter, and sample-return missions. Several missions were analyzed assuming two flight options--ballistic and SEP low-thrust. The performance results of this subtask were subsequently used as inputs to Shuttle/IUS evaluation analyses performed by Code SL prior to the IUS concept selection.

2.1.3 Mars Penetrator Mission

A short paper describing the Mars penetrator mission was presented at the Fifth Annual Planetology Program Principal Investigators' (PPPI) Meeting at the request of Code SL. Primary emphasis of the paper was on exploratory science capability of the penetrator concept. The advantages of simple design, multiple sites and lower cost were presented against the disadvantages of severe environment and limited capability. A summary of the details of the penetrator design, mission alternatives, operations profiles, and penetrability characteristics were also discussed.

2.1.4 Escape Performance Envelope of IUS Candidates

The purpose of this task was to represent the quoted escape performance estimates of the five IUS concepts on a common format of injected payload versus vis viva energy (C3). This effort included plotting the various sub-options within each IUS Candidate designed to address planetary mission requirements. These results were compared against the Titan IIIE/Centaur/TE364-4 launch vehicle to illustrate changes in performance of all possible Shuttle/IUS configurations for interim-period missions. Finally,

TABLE 1

Summary of 1975-76 Advanced Planning Activity

Subtask	Dates	Subject Title	Submitted To
1	Feb. 1975	Mars Penetrator Design Status Report	COMPLEX/SSB
2	Mar. 1975	1975 Planetary Mission Model Performance Requirements	Code SL/NASA
3	Mar. 1975	Mars Penetrator Mission	5th Annual PPPI Mtg.
4	Mar.-Apr. 1975	Escape Performance Envelope of IUS Candidates	Code SL/NASA
5	Apr. 1975	Flight Time Performance of 1980 50-AU Solar Apex Mission	Owen/SUNY
6	May 1975	Performance Summary of 1980-83 Pioneer Jupiter Swingby Execliptic Missions	Code SL/NASA
7	May 1975	1980 and 1981/2 Minimum PJ _p Transfers	Code SL/NASA
8	May 1975	Cost Assessment of Candidate Centaur IUS Options for Planetary Missions	Code SL/NASA
9	May-Dec. 1975	1975 Comet Working Group Technical/Editorial Support	1975 CWG
10	June 1975	Viewgraph Preparations for D. Herman COMPLEX Presentations	Code SL/NASA
11	June-July 1975	Status Report on Mission Concepts for Planetary Surface Exploration	COMPLEX/SSB
12	July 1975	1980 Encke/Mercury Flyby Mission Transfer Analysis	Code SL/NASA
13	July 1975	Pioneer Venus Follow-On Mission Opportunity Characteristics	Code SL/NASA
14	July 1975	Summary of Execliptic Mission Options	Simpson/U of C
15	Aug. 1975	Quick-Look Evaluation of Candidate Shuttle/IUS Launch Systems Performance	Code SL/NASA
16	Aug. 1975	Quick-Look Performance Evaluation of Solid IUS Candidates with SEP for Planetary Missions	Code SL/NASA
17	Aug. 1975	Saturn Orbiter Mission Performance Assessment Using Shuttle/IUS (Solids) and SEP Systems	Code SL/NASA

TABLE 1 (Continued)

Summary of 1975-76 Advanced Planning Activity

Subtask	Dates	Subject Title	Submitted To
18	Aug. 1975	Saturn Orbiter Trip Time Sensitivity to SEP Thrust Time	Code SL/NASA
19	Aug. 1975	Panel Discussion Moderator: Can STS Improve the Planetary Program	AAS 21st Mtg.
20	Sept. 1975	Launch Vehicle Alternatives for Transition Period Missions	Code SL/NASA
21	Sept. 1975	Flight Time Requirements for Saturn Orbiter Missions	Code SL/NASA
22	Sept. 1975	Planetary Mission Options Summary: 1979-1985	Code SL/NASA
23	Sept. 1975	Solid IUS Planetary Applications Summary	ARC/NASA
24	Sept. 1975	Parametric Data for 1986 SEP Comet Tempel-2 Rendezvous Mission	Cork/JPL
25	Sept. 1975	Solid IUS Summary and Planetary Missions Capture Matrix	COMPLEX/SSB
26	Oct. 1975	Planetary Mission Model Support Data	Cork/JPL
27	Oct. 1975	Injection Energy (C3) Summary of Saturn/Uranus Swingby and Uranus Direct Flyby Missions with Constant Six-Year Trip Time	Code SL/NASA
28	Dec. 1975	COMPLEX Report Editorial Assistance	Code SL/NASA
29	Jan. 1975	Uranus Flyby Performance Summary	Code SL/NASA

an envelope of the possibilities was prepared and again plotted on a format of injected payload versus C3. The results of Subtask 2 were then superimposed to determine where the requirements of the current planetary mission model were located with respect to this envelope. These plots proved to be effective comparisons contributing to the formation of a Planetary Programs position regarding Candidate IUS preferences.

2.1.5 Flight Time Performance of a 1980 50-AU Solar Apex Mission

The payload versus trip time performance trade-off was investigated for a 50-AU Solar Apex Mission. Since minimum time to reach interstellar space was the primary performance criterion, the 1980 Jupiter swingby opportunity, which provides favorable solar apex geometry, was selected for the analysis. As a bound on performance, the smallest and largest launch vehicles considered were the Atlas/Centaur/TE364-4 and the Saturn V/Centaur/TE364-4. Performance results yielded payload limits for the 50-AU mission of 300 kg and 25,000 kg for these two vehicles, respectively. At the 300 kg level the trip time required was shortened from more than 20 years to 6 years by using the Saturn V launch vehicle configuration.

2.1.6 Performance Summary of 1980-83 Pioneer Jupiter Swingby Execliptic Missions

A brief performance analysis was conducted to compare the heliographic latitudes attainable by Pioneer spacecraft on Jupiter swingby trajectories launched by several candidate launch vehicles. The spacecraft mass was parameterized from 210 to 260 kg, with an additional payload of 650 kg also considered to represent a dual Pioneer spacecraft launch. The 1980, '81/2, and '83 Jupiter swingby launch opportunities were selected for the analysis as compatible with the planetary mission model. Five launch vehicles, beginning with the Atlas/Centaur/TE364-4, were analyzed. The remaining four vehicles were Shuttle launches with Candidate IUS escape stages. Results obtained show that the 1983 opportunity provides the best heliographic latitudes performance for any given vehicle/payload combination. The Atlas/Centaur/TE364-4 performance provides latitudes of about 40° to 70°

for a single launch, depending upon the payload and launch year. The largest vehicles considered, the IUS 25' and 31' Expendable Centaurs easily deliver a single spacecraft to $>85^\circ$ in any year. For a dual launch, the highest latitude attained was 79° with the Shuttle/IUS '31 Centaur and a 1983 launch.

2.1.7 1980 and '81/2 Minimum PJ_p Transfers

A brief analysis of minimum energy Jupiter transfers was performed to ascertain if the maximum deliverable payload of the Atlas/Centaur/TE364-4 was sufficient for a Pioneer Jupiter Probe (PJ_p) mission. Three different transfers were analyzed: 1) minimum energy 1980 Type II transfers with a 7-day launch window, 2) minimum energy 1982 Type II transfers with a 5-day launch window, and 3) 1981-2 minimum energy Mars gravity-assisted Jupiter transfers with a 15-day window. In all three cases, the transfer flight times were nearly 1075 days (2.9 years). The injected payloads the Atlas/Centaur/TE364-4 can deliver on these transfers are 295, 325, and 310 kg, respectively. These values are well below the 425 to 475 kg range nominally considered for a baseline PJ_p mission. Although an independent design analysis conducted for ARC/NASA indicated that a 300 kg PJ_p mission might be feasible, it required use of all available performance margins with no mass contingency. Hence, it was concluded that an Atlas/Centaur/TE364-4 launched PJ_p mission was too high a risk to be an attractive planning option.

2.1.8 Cost Assessment of Candidate Centaur IUS Options for Planetary Missions

The purpose of this task was to investigate the planetary transportation (launch vehicle) costs for the period 1980-90 of several IUS Centaur introduction scenarios. Four vehicle configuration cases were considered in the analysis: Case 1) All Shuttle launches use the Expendable IUS Centaur; Case 2) All Shuttle launches use the Recoverable IUS Centaur; Case 3) All Shuttle launches before 1985 use the Expendable IUS Centaur; those after 1984 use the Recoverable IUS Centaur; and Case 4) same as

Case 3 except the Space Tug is used instead of the Recoverable IUS Centaur after 1984. The rate of introduction of these Shuttle-based vehicle Cases was controlled by three subcases within each Case. Given the eight planetary missions of the mission model between 1980 and 1984, these subcases were defined as: Subcase a) All eight missions are launched with expendable vehicles, i.e., the first Shuttle launch occurs in 1985; Subcase b) Four missions are expendable launches and four missions are Shuttle launches, and Subcase c) All eight missions are launched with Shuttle-based vehicles. The final results showed that there was little difference in the total transportation cost of the four Cases but that the cost variation between the Subcases, i.e., rate of Shuttle-based vehicle introduction, was significant. About a 25 percent savings was realized in each Case by going from Subcase a to Subcase b. Using the fastest rate of introduction, Subcase c, provided about an additional 15 percent savings in all Cases.

2.1.9 1975 Comet Working Group Technical and Editorial Support

The purpose of this assignment was to provide technical and editorial support to the 1975 Comet Working Group which met May 12-13, 1975 to compare the merits of the 1980 Mariner Encke flyby opportunity with the 1981 Explorer Multi-Comet flyby opportunity as first comet missions. The Working Group concluded that a 1980 Encke mission with the flyby encounter at ≤ 0.4 AU was most preferred. Individual written contributions in support of this conclusion were collected and organized with additional technical data into a meeting report entitled "On the Choice of a First Comet Mission." The original draft was circulated among the Committee members for review and a revised draft was submitted to the COMPLEX Summer Study meetings at Seattle in July. Quantitative flyby imaging data was subsequently added, followed by a second review and the report was published in the fall of 1975.

2.1.10 Viewgraph Preparations for D. Herman COMPLEX Presentations

Two sets of viewgraphs were prepared at the direction of

Dan Herman for presentations requested by COMPLEX on the subjects:

1) general statement on current stage of inner planet explorations, and
2) outer planet mission options for the next decade. Eight viewgraphs were prepared for the inner planets presentation. The subjects covered included the current inner planets mission model, Mercury orbiter opportunities of the 80's, multi-Venus swingby Mercury orbiter flight profile, Mars mission performance summary, Jupiter swingby exocentric mission performance, 1980 Encke flyby mission options, and flight profiles of a 1980 VEGA multi-asteroid survey mission and a 1986 Tempel-2 SEP rendezvous mission. Additional figures on the VOIR mission were provided by JPL. Five viewgraphs were submitted for the outer planets opportunities presentation. Included were the current outer planets mission model, launch requirements summary of outer planet flybys, summary of Saturn/Uranus flyby mission options, flight time summary of outermost planet missions, and minimum energy summary for Jupiter and Saturn Orbiter missions. Two additional graphs were supplied by JPL on Jupiter orbiter and/or probe mission options in the early-80's.

2.1.11 Status Report on Mission Concepts for Planetary Surface Exploration

At the request of the COMPLEX, a status report was prepared for their Summer Study activity on planetary surface exploration concepts and related SR&T efforts. The report addressed: 1) the Mars penetrator mission, 2) other inner planet surface missions, and 3) outer planet satellite missions. The penetrator concept was most heavily emphasized in all three sections, being the most viable alternative to large expensive soft landers that is currently under consideration. For the Mars penetrator mission the penetrator design, deployment options, mission objectives, and SR&T activities in soil modifications and instrument component qualifications were discussed. For other inner planet surface missions, the Venus and Mercury penetrators were briefly described along with a description of an Alternate Lander Study just initiated at JPL. For the outer planet surface missions, primary consideration was given to the definition of Galilean satellite penetrators. Atmospheric/surface missions to Titan were also

introduced. As part of this support activity for COMPLEX, three presentations covering the material in this report were given to the Committee during their Seattle meeting when the report was submitted.

2.1.12 1980 Encke/Mercury Flyby Mission Transfer Analysis

The recommendation of the 1975 Comet Working Group for a 1980 Encke flyby mission at ≤ 0.4 Au was considered by the COMPLEX in the mission strategy activity during their 1975 summer meeting in Seattle. In an effort to enhance the capability and science objectives of such a mission, the suggestion was made to add a flyby of Mercury as a second encounter. The purpose of this analysis was to examine the trajectory and performance requirements of a combined 1980 Encke/Mercury flyby mission. A brief study of the problem revealed several possible solutions. A launch window analysis was then performed on the most favorable of these possibilities. It was concluded that sufficient net payload would be available to a Mariner 1980 Encke/Mercury mission if the Titan/Centaur launch vehicle were possible. Earth-storable retro propulsion would have to be added to perform a significant midcourse retargeting maneuver of more than 1 km/sec following the Encke encounter in order to reach Mercury. The flyby speeds at Encke and Mercury would be 16 and 19 km/sec, respectively, somewhat higher than typically desired values. Other possible opportunities for combining Encke and Mercury encounters on a single mission are probably available, including additional inner planet swingbys, but no further effort was requested to extend the search for such trajectories.

2.1.13 Pioneer Venus Follow-On Mission Opportunity Characteristics

A brief summary of the transfer characteristics of the 1980, 81, 83 and 85 launch opportunities to Venus was performed for Code SL in order to assess the systems implications of a delayed launch of the Pioneer Venus spacecraft. The analysis was required in response to a congressional interest in delaying the completion of the Pioneer Venus project. Each opportunity had to be analyzed for the correct combination of trajectories

which would minimize energy, and have acceptable arrival dates for the combined operations of the probe and orbiter spacecraft. Arrival velocities also had to be controlled to minimize systems changes on both the probes and the orbiter. Even so, it was determined that it would be necessary to replace the orbiter retro insertion motor with a larger unit for the 1980 opportunity due to uncontrollably higher approach velocities. This was considered a significant reason for not slipping the launch schedule of Pioneer Venus from 1978 to 1980. To slip the schedule further would not impact the systems requirements but would surely destroy the cost effectiveness of this program.

2.1.14 Summary of Execliptic Mission Options

A survey of execliptic mission options was conducted and the results summarized in support of a theme paper on execliptic mission options by Dr. John Simpson of the University of Chicago. The analysis was performed in response to his request of the Planetary Programs Division, NASA Headquarters for a definitive description of all possible execliptic mission options under consideration. Ten options were prepared involving direct, Jupiter swingby and VEGA trajectories. Both ballistic and solar electric low thrust flight modes were addressed. Launch vehicles considered included the Delta 2914, Delta 3914, Delta 3914/Star-27, Atlas/Centaur, Atlas/Centaur/TE 364-4, Titan IIIE/Centaur, Titan IIIE/Centaur/TE 364-4, and Shuttle/Transtage/TE 364-4. Solar electric propulsion power levels of 6 kw and 15 kw were chosen. The options included single and dual spacecraft launches. Performance, in terms of attainable heliographic latitude, varied from as low as 30° with a single spacecraft to as high as 90° with dual spacecraft. Flight times were as short as 75 days for a direct ballistic launch to low latitudes to as long as 4 years for Jupiter swingby missions of dual spacecraft to polar heliographic latitudes.

2.1.15 Quick-Look Evaluation of Candidate Shuttle/IUS Launch Systems Performance

An evaluation was requested for a performance comparison between a Shuttle-launched two-stage solid IUS with a 21 kw SEP Stage and a Shuttle-launched Centaur D1-S with a TE 364-4 kick stage. Specifically of interest was the question, given a two-stage solid IUS: "Should the Planetary Program develop an SEP upper stage for this IUS or should it independently meet its mission performance requirements with a separately developed Centaur D1-S/TE 364-4?" Because it was particularly relevant to this question, a third alternative was added to the analysis; it being a three-stage solid IUS having two large first stages in tandem (fired serially) and the same upper stage as the standard two-stage configuration.

Each launch vehicle option was applied to the current planetary mission model. It was found that the IUS/SEP option was least satisfactory due to insufficient energy in the two-stage IUS to deliver the SEP stage and spacecraft payload to sufficient escape velocity. The Centaur D1-S/TE 364-4 performed best but was considered most expensive and probably least compatible with Shuttle operational desires. It was recommended that the three-stage IUS be given further considerations as a baseline launch system for transition period planetary missions.

2.1.16 Quick-Look Performance Evaluation of Solid IUS Candidates with SEP for Planetary Missions

In response to the recommendations of Subtask 2.1.15, NASA Headquarters requested a comparison of payload performance between the following three launch vehicles:

- 1) Shuttle/Centaur D1-S/TE 364-4
- 2) Shuttle/3-Stage BII/SEP(21 kw)
- 3) Shuttle/3-Stage BII/TE 364-11

for a Mercury and Jupiter orbiter missions. The 3-Stage BII is the same configuration as the three-stage IUS suggested above. Net orbited Mercury payload was presented for each of these launch vehicles as a function of

orbit eccentricity for a constant 500 km periapse altitude orbit. Solid retro orbit insertion was assumed. The first launch vehicle configuration provided the best performance but required 950 days to reach Mercury using the assumed 1983 triple-Venus swingby opportunity. The second configuration using SEP provided less than 20 percent less payload and only required 350 days to reach Mercury. For the Jupiter orbiter mission both the 1985 (bad) and 1987 (good) opportunities were considered. A Ganymede-assisted orbit with a periapse radius of $13.2 R_J$ and a period of 100 days was assumed. The second vehicle configuration using SEP was best providing over 1200 kg net orbited payload with an earth-storable retro propulsion system.

2.1.17 Saturn Orbiter Mission Performance Assessment Using Shuttle/IUS (Solids) and SEP Systems

This subtask was another extension of the analysis performed in the two previous subtasks to determine the performance capability of emerging Shuttle-based launch systems applied to high-energy planetary missions. In particular, the purpose of this effort was to analyze the ability of a SEP thrust module to successfully improve several Shuttle-based upper stage configurations to a point where Saturn orbiter missions could be performed with sufficient net payload and reasonable flight times (<5 years). Shuttle/IUS(3) and Shuttle/Centaur(R) launch vehicles were considered and earth-storable retro propulsion was assumed. The preferred capture orbit assuming Titan-assist had a periapse of $19.7 R_S$ and a period of 95.67 days (6:1 resonance with Titan for subsequent orbit pumping and cranking). At a 5-year trip time, net payload was limited to 580 kg with the IUS(3), but increased to 1115 kg with the Centaur(R) used in an expendable mode. At comparable payload the Centaur(R) option, again expended, shortened the trip time to Saturn to 4.1 years.

2.1.18 Saturn Orbiter Trip Time Sensitivity to SEP Thrust Time

The results presented in Subtask 2.1.17 assumed a fixed SEP thrust time of 350 days. In an effort to shorten the flight time of the

IUS(3)/SEP (25/21 kw) configuration, a request was made to investigate longer thrust times. A plot was prepared of flight time versus propulsion on-time for a fixed net orbited mass of 675 kg. The results show that tripling the propulsion on-time to 1050 days only shortens the flight time by 26 days from 1974 days to 1948 days, just a little more than 1 percent reduction. The reason is, of course, that the SEP module is so far from the Sun after 350 days that there is little power still available to continue operating the thrusters.

2.1.19 Panel Discussion Moderator: Can STS Improve the Planetary Program?

An Automated Spacecraft-Planetary Session entitled "Can We Use the STS to Improve the Planetary Program?" was held as part of the 21st Annual AAS Meeting addressing "Space Shuttle Missions of the 80's" in Denver, Colorado, 26-28 August 1975. Mr. John Niehoff of Science Applications, Inc. was requested, with NASA Headquarters approval, to moderate a panel discussion of this topic as the second part of the session. Participants on the panel included Messrs. D. Herman (OSS/NASA HQ), J. Wild (OMSF/NASA HQ), R. Parks (JPL), and P. Culbertson (OMSF/NASA HQ). As an introduction to the panel discussion, Mr. Niehoff gave a short paper on anticipated launch vehicle costs for the planetary mission model assuming three different Shuttle upper stage introduction scenarios. These scenarios were:

- a) Use of the IUS with necessary performance modifications;
- b) Use of the Centaur D1-T irregardless of the IUS choice; and
- c) Use of a low-technology Tug instead of the IUS.

In each case, the impact of a subsequent introduction of the recoverable high-technology Tug in 1985 was also considered. The results showed that an early and complete transition to Shuttle-launched planetary missions will be far more significant than the cost benefits of any of these specific Shuttle-based scenarios. Also, the introduction in 1985 of a high-technology Tug does not provide significant additional economies for planetary missions.

2.1.20 Launch Vehicle Alternatives for Transition Period Missions

A brief investigation of launch vehicle alternatives was performed for 1980 and 1981 planetary missions in the event of a delay in bringing the Space Transportation System (STS, i.e., Shuttle) to operational status. Four missions were considered: 1) 1980 Pioneer Jupiter Execliptic, 2) 1980 Mariner Encke Flyby, 3) 1980 and 1981 Pioneer Jupiter Orbiter with Probe, and 4) 1981 Pioneer Mars Penetrator. Three launch vehicles were reviewed: 1) Atlas/Centaur, 2) Titan IIIE/C, and 3) Titan IIIC. The TE 364-4 Kick Motor was added to these vehicles for high energy missions. A tabular summary comparing required injection conditions with the vehicle capabilities was used to present the results. The Atlas/Centaur successfully captures missions No. 1 and 4. The Titan IIIE/Centaur captures all four missions. The Titan IIIC captures missions No. 1, 2 and 4 with considerable margin on mission No. 4.

2.1.21 Flight Time Requirements for Saturn Orbiter Missions

The flight time requirements of Saturn Orbiter Missions using a range of launch vehicles and flight modes was investigated. Assumed were a Titan-assisted orbit insertion, earth-storable retro propulsion, and a 675 kg net orbited payload. Five launch configurations were examined including:

- 1) Burner IUS(3)/TE 364-11
- 2) Burner IUS(3), VEGA
- 3) Burner IUS(3)/TE 364-11, ΔVEGA
- 4) Burner IUS(3)/SEP(25/21 kw)
- 5) Centaur (Reusable/Expendable)

Flight time to Saturn varied from impossible (insufficient energy to reach Saturn's orbit) for the first configuration, to 4.8 years for the last configuration. Additional graphs parameterizing injected mass and net orbited mass as a function of escape energy (C3) were also prepared as back-up material.

2.1.22 Planetary Mission Options Summary: 1979-1985

The purpose of this subtask was to summarize the performance margins available on several transition period (1979-1985) planetary missions launched with either the Titan IIIE/Centaur D1-T or Shuttle-based/IUS vehicles. The missions considered included:

- 1) Mariner Jupiter Orbiter,
- 2) Pioneer Jupiter Orbiter with probe,
- 3) Viking Follow-On,
- 4) Pioneer Jupiter Probe,
- 5) Mariner Saturn/Uranus Flyby.

SEP low-thrust as well as ballistic transfer modes were analyzed. Results were presented in tabular form along with detailed summaries of relevant transfer data for future advanced studies use. These data were provided as general supporting material for planning purposes, hence, no subtask conclusion or recommendations were drawn.

2.1.23 Solid IUS Planetary Applications Summary

A definition/performance summary presentation was prepared for ARC/NASA management of Shuttle-based Burner II IUS concepts for Pioneer planetary missions. The presentation scope included three areas of interest: 1) definition of pertinent launch/upper stage definitions, 2) summary of Pioneer planetary mission performance with these definitions, and 3) the status and future milestones of the IUS. The presentation was given to the ARC Director and upper management along with a second presentation concerning Shuttle launch environment implications for planetary spacecraft. Considerable discussion followed these presentations regarding initial Shuttle operational dates and capabilities. A brochure of the presentation material was also prepared and distributed at the meeting.

2.1.24 Parametric Data for 1986 SEP Comet Tempel-2 Rendezvous Mission

Parametric payload performance data were generated for a 1986

launched SEP rendezvous mission with comet Tempel-2. The data were prepared as a check against previous planetary mission model capture analyses of IUS candidates performed at JPL. Plots of net mass versus initial mass and net mass versus initial power were generated to determine maximum performance with two solid IUS two-stage configurations. The results were found to be compatible with previous results. However, considerably more margin was found to be available than previously indicated, i.e., 840 kg capability versus a 450 kg requirement.

2.1.25 Solid IUS Summary and Planetary Missions Capture Matrix

At the direction of the NASA Headquarters, in response to a request by the COMPLEX, a brief presentation was given defining the capabilities of the solid IUS configurations for Shuttle-launched escape missions. As a follow-up to this presentation a matrix of launch vehicle scenarios was prepared at the meeting to indicate which candidate planetary missions were most vulnerable to changes in the current Shuttle development plan. A total of 8 launch vehicle scenarios and 28 planetary mission opportunities were considered for the period 1979 to 1988. The most preferred launch vehicle development scenario from the standpoint of mission capture success was:

- a) Atlas/Centaur/TE 364-4 until Shuttle/IUS IOC,
- b) Shuttle/IUS(3)/TE 364-4-11 IOC by 1981, and
- c) SEP(25 kw) IOC by 1984.

A high-technology Tug by 1987 without SEP was not considered as favorable from the mission capture viewpoint.

2.1.26 Planetary Mission Model Support Data

This subtask is an extension of the supporting data developed in Subtask 2.1.22 above. In particular, it was requested that transfer characteristics, mass summaries, and orbit data (where appropriate) be provided for the following specific missions:

- 1) 1981 Pioneer Mars Penetrator,
- 2) 1981/2 Pioneer Jupiter Orbiter/Probe,
- 3) 1981/2 Pioneer Jupiter Execliptic,
- 4) 1982, 83/4, 85 Pioneer SUT_p Swingby.

These data were generated along with Systems Study Report references and sent to JPL in support of their activity regarding mission model definition for Shuttle/IUS systems study applications.

2.1.27 Injection Energy (C3) Summary of Saturn/Uranus Swingby and Uranus Direct Flyby Missions with Constant Six-Year Trip Time

A specific request for C3 variations of 10-day launch windows for Saturn/Uranus swingby opportunities was received for the four launch opportunities from 1981/2 to 1985. Trajectory data were rerun for these cases to provide accurate results. The C3's were found and transmitted along with comparative C3's for direct ballistic Uranus missions for the same opportunities. These results were forwarded to COMPLEX in response to a request for maximum C3's of the current planetary mission model.

2.1.28 COMPLEX Report Editorial Assistance

Technical editing support was provided for COMPLEX's final report preparation at the direction of NASA Headquarters. Specific assistance was given regarding the subject of launch vehicle performance capabilities and Shuttle-based IUS availability. This assistance included verification of launch vehicle capability design points (injected payload and C3) to be used as specification of recommended capability.

2.1.29 Uranus Flyby Performance Summary

A performance summary of Uranus flyby mission opportunities remaining in this century was prepared. A graphical format of net flyby payload versus trip time was chosen to compare the various flight modes and opportunities. Assumed as constant in the comparison were a 10-day launch window and the Shuttle/IUS(3)/TE 364-11 launch vehicle. Four graphs were prepared:

- 1) Uranus flyby performance comparison for 1979-81 Jupiter swingbys and direct missions;
- 2) Uranus flyby performance comparison for 1980-85 Saturn swingbys and direct missions;
- 3) Uranus flyby performance comparison for 1992-95 Jupiter swingbys and direct missions;
- 4) Uranus flyby performance comparison for 1980-1994 direct missions.

These data graphically illustrate the performance superiority of Jupiter swingby opportunities to Saturn swingbys, and the superiority of Saturn swingby opportunities to direct missions. There is very little variation in ballistic direct mission performance over the annual opportunities from 1980 to 1994.

2.2 Cost Estimation Research (1604 Man-Hours)

This task is continuing research in the development of a planetary mission cost estimating model; the purpose of this model being to provide estimates of future planetary missions for the Planetary Programs Division of NASA. Historically, cost estimates of future missions have been in error by extreme amounts and have been related to the complexity of the proposed mission. Two apparent reasons for these inadequate estimates are: 1) the interfacing of new technology has not been appreciated, and 2) the use of inadequate data bases for cost modeling.

The cost model which has been developed and refined under this contract has as a goal an accuracy level of ± 25 percent on the estimate. The scope of the model covers a wide range of planetary mission types including flybys, orbiters, landers, probes, penetrators, and sample returns. The model input requirements have been limited by the requirement that they be available at the time of the pre-Phase A level definition. A list of the input requirements is shown in Figure 1.

A functional summary of the cost model is shown in Figure 2. As can be seen from Figure 2, the basic estimating unit is man-hours. Planetary missions can be characterized by very low production volume and very high development costs. In this mode of operation the key element of cost is man-hours. Also, the common denominator of the NASA cost reporting system is the cost incurred in direct labor, since from this base the overhead, G & A and fee are computed. The other category of costs (ODC's) is significantly smaller than the charge for labor and is only of secondary importance.

Estimating manpower, rather than dollars, has the following benefits: 1) simplifying the actual estimation procedure since fewer cost elements are involved; 2) removing the effects of inflation from the estimating procedure; and 3) providing added visibility to the cost reduction effects of learning and inheritance.

The basic estimation of the model is done at the subsystem hardware level. It starts by estimating the non-recurring direct labor hours

INPUTS FOR COST MODEL

- ③ DATE OF FIRST LAUNCH (Z\$)
- ③ FISCAL WAGE DATE (D2)
- ③ NUMBER OF FLIGHT ARTICLES (N1)
- ③ WEIGHT OF POWER SUBSYSTEM EXCLUDING RTG'S (W1)
- ③ NUMBER OF RTG UNITS PER SPACECRAFT (N2)
- ③ RTG FUEL LOADING (THERMAL WATTS) (L1)
- ③ TOTAL WEIGHT OF STRUCTURE SUBSYSTEM (S1)
- ③ WEIGHT OF MECHANISMS AND LANDING GEAR (S2)
- ③ WEIGHT OF THERMAL CONTROL, PYRO, AND CABLING (S3)
- ③ PROPULSION SYSTEM DRY WEIGHT (EXCLUDING THROTTLEABLE LIQUID VERNIER FOR LANDERS) (P1)
- ③ LIQUID VERNIER DRY WEIGHT (P2)
- ③ AERODECELERATION SUBSYSTEM WEIGHT (P3)
- ③ TOTAL WEIGHT OF GUIDANCE/CONTROL SUBSYSTEM (G1)
- ③ WEIGHT OF RADAR IN G/C SUBSYSTEM (G2)
- ③ WEIGHT OF RADIO FREQUENCY COMMUNICATIONS SUBSYSTEM (C1)
- ③ WEIGHT OF DATA HANDLING SUBSYSTEM (C2)
- ③ DIAMETER OF ANTENNAS (C3)
- ③ TOTAL WEIGHT OF SCIENCE EXPERIMENTS (Q1)
- ③ WEIGHT OF LANDER SURFACE EXPERIMENTS (HAVING SIGNIFICANT SAMPLING/PROCESSING OPERATIONS) (Q2)
- ③ PIXELS PER LINE OF TV (Q3)
- ③ TOTAL MISSION DURATION (K1)
- ③ TOTAL ENCOUNTER TIME (K2)
- ③ NUMBER OF LAUNCH WINDOWS (K3)

FIG. 1

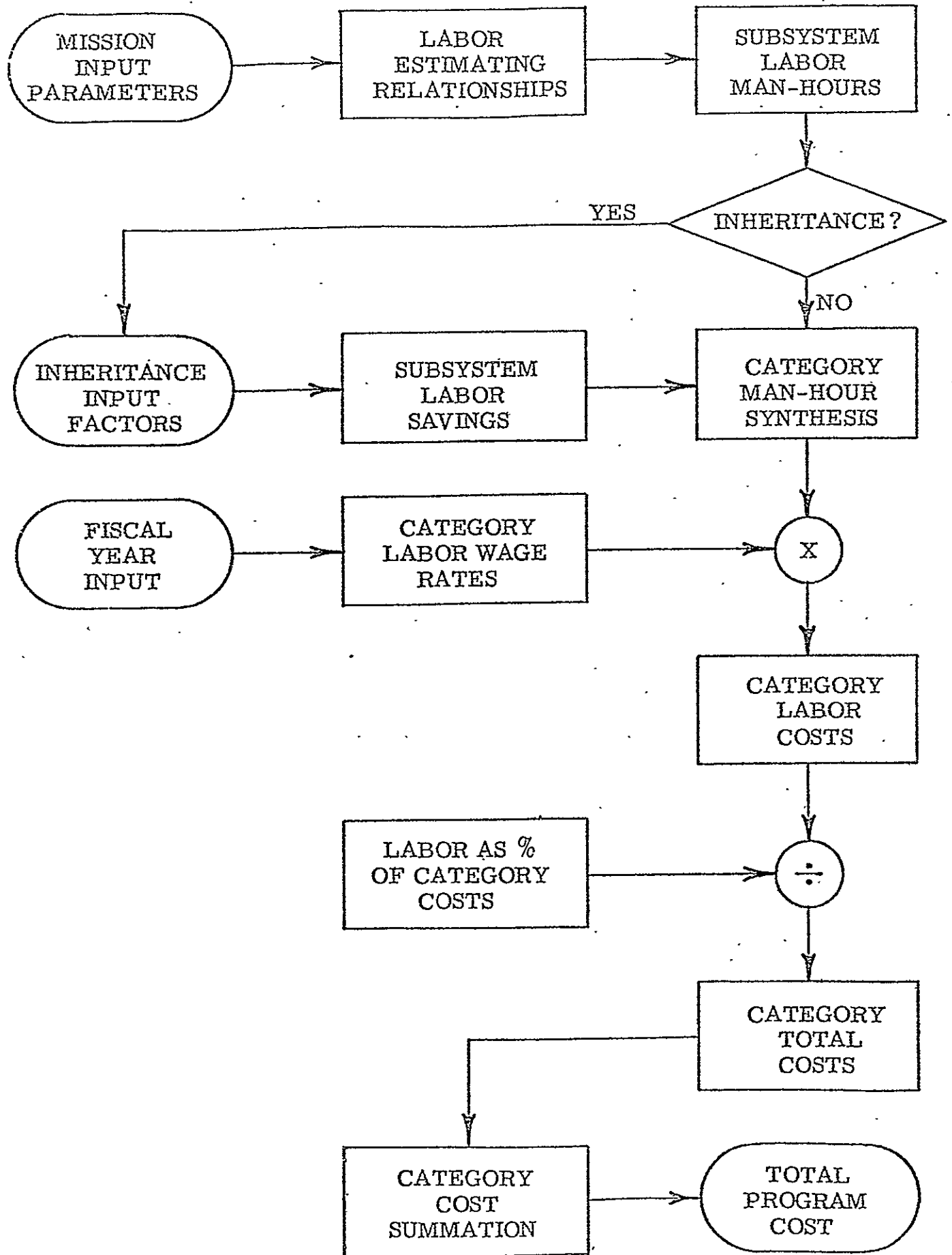


FIG. 2 - PLANETARY COST MODEL SCHEMATIC

for the hardware subsystems with subsequent estimates of support functions and costs being built upon these values. The Labor Estimating Relationships (LER's) provide resolution down to the component hardware level. However, the LER's begin at the subsystem level so that the input requirements do not exceed the information content of pre-Phase A mission studies. The inputs are composed largely of subsystem weights and key mission events due to this requirement.

The model also has the ability to factor in the cost benefits of direct inheritance from previous projects utilizing similar or identical subsystems. The inheritance modeling is applicable to both hardware and design inheritance and is general enough to permit the inclusion of cost benefits from standardized hardware at some future date.

The data base of the SAI planetary cost model is constantly being updated and expanded. During the past year three missions were essentially completed. These were the Mariner Venus/Mercury, Viking Lander and the Viking Orbiter missions. The Mariner Jupiter/Saturn mission is now 35 percent complete in the data base with an updated estimate to completion. The distribution of cost and labor in the updated base is shown in Figures 3 and 4. This data reflects a continued stability in the distribution of labor and of costs.

With the large quantity of new data replacing the estimates to completion, one of the major tasks during the past year was a major refit of all of the LER's using the new data. The new LER's which were derived from the refit are shown in Figure 5. All of the LER's had minor changes in form and coefficients except for the Communications, Launch and Flight Operations, and Data Analysis categories which had major revision.

The present cost model has been applied to planetary missions scheduled through the 1980's and to several advanced planning activities. The results have been very encouraging and plans are to continue with more effort in the applications area. An example of one application is shown in Figures 6 and 7 and Table 2 in which five options of one mission were modeled. The conclusions reached from this exercise were: 1) The

cost model is effective in analyzing mission option levels, 2) cost trade-off analysis require fairly detailed advanced study results, 3) mission operations costs will be high for long duration outer planet missions, unless new approaches are developed, and 4) inheritance and hardware standardization will impact systems contractor costs the most; spacecraft costs might be reduced by a third or more. The cost estimation research is currently being continued to expand the data base and add capability to estimate more ambitious projects.

LABOR HOURS DISTRIBUTION COMPARISON

• SUPPORT CATEGORIES

- o PROGRAM MANAGEMENT
- o SYSTEMS ANALYSIS
- o TEST
- o QUALITY ASSURANCE AND RELIABILITY
- o ASSEMBLY AND INTEGRATION
- o GROUND SUPPORT EQUIPMENT
- o LAUNCH AND FLIGHT OPERATIONS
- o DATA ANALYSIS

• SUBSYSTEM CATEGORIES

- o STRUCTURE
- o PROPULSION AND AERODECELERATION
- o GUIDANCE AND CONTROL
- o COMMUNICATION
- o POWER
- o SCIENCE

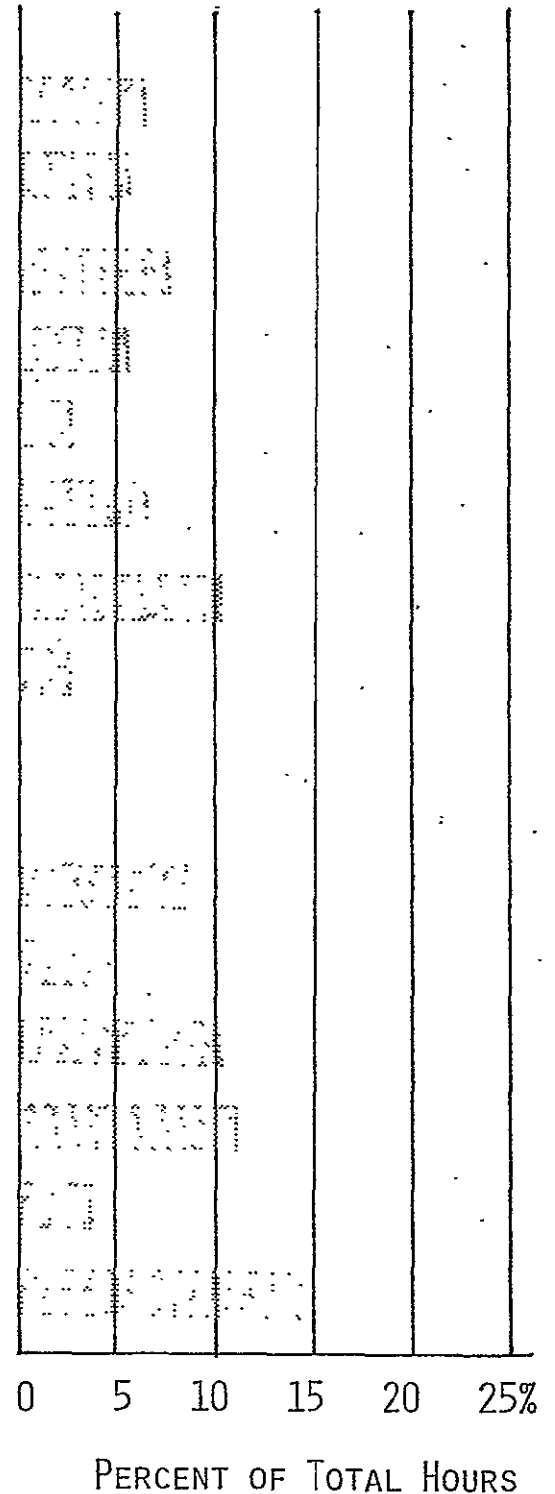


FIG. 3

COST DISTRIBUTION COMPARISON

● SUPPORT CATEGORIES

- PROGRAM MANAGEMENT
- SYSTEMS ANALYSIS
- TEST
- QUALITY ASSURANCE AND RELIABILITY
- ASSEMBLY AND INTEGRATION
- GROUND SUPPORT EQUIPMENT
- LAUNCH AND FLIGHT OPERATIONS
- DATA ANALYSIS

● SUBSYSTEM CATEGORIES

- STRUCTURE
- PROPULSION AND AERODECELERATION
- GUIDANCE AND CONTROL
- COMMUNICATION
- POWER
- SCIENCE

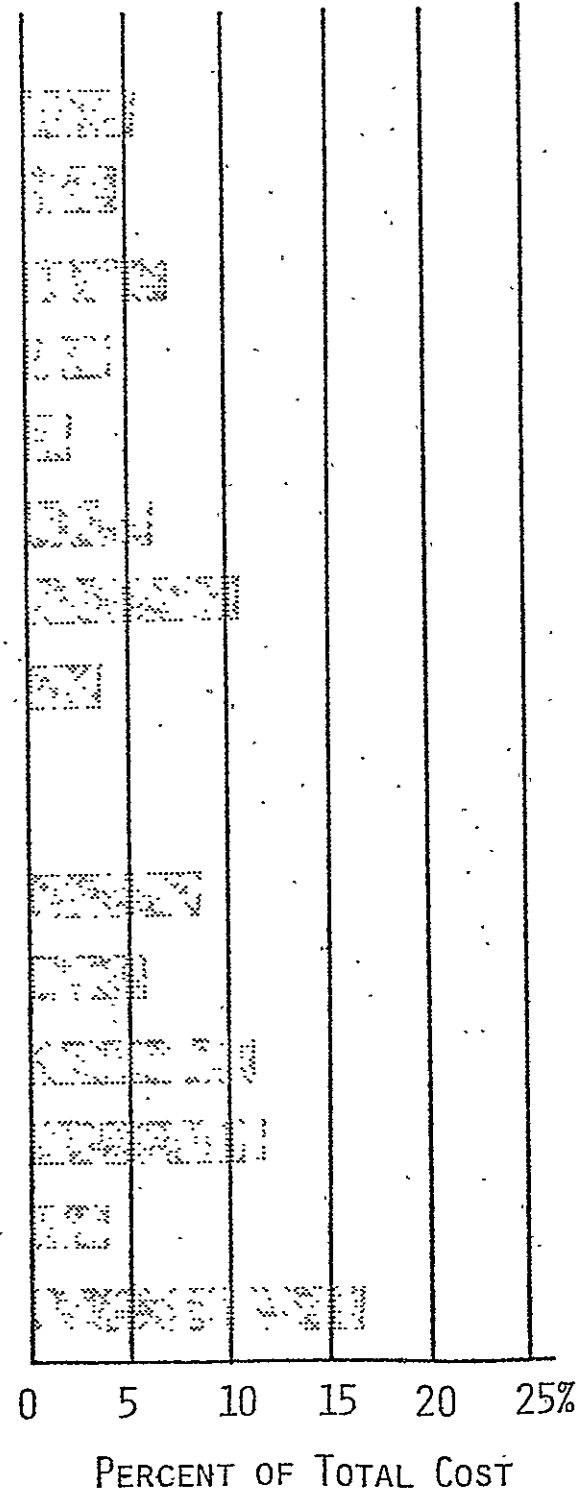


FIG. 4

COST ESTIMATION MODEL LER's

$$\bullet \text{ NR}_{ST} = 3.98(S2)^{0.842} + 37.20(S3)^{0.341} + 21.85(S1-S2-S3)^{0.426}$$

$$\bullet \text{ NR}_P = 17.75(P1)^{0.527} + 7.9(P2)^{0.88} + 19.5(P3)^{0.5}$$

$$\bullet \text{ NR}_{GC} = 47.99(G1)^{0.499} + 10.0(G2)^{1.24}$$

$$\bullet \text{ NR}_C = 5.09(C1) + 58.61(C2)^{0.415} + 28.75(C3)$$

$$\bullet \text{ NR}_{EP} = 0.48(W1) + 177$$

$$\bullet \text{ NR}_{SE} = 3.8(Q1) + 6.0(Q2)^{1.25} + 0.16(Q3) + 2.2$$

$$\bullet \text{ DLH}_{PM} = 0.067(\text{DLH}_{SS})^{1.04}$$

$$\bullet \text{ DLH}_{AI} = 0.15(\text{DLH}_{SS})^{0.83}$$

$$\bullet \text{ DLH}_{GE} = 0.033(\text{DLH}_{SS} - \text{DLH}_{ST})^{1.1} / (1 - 0.7 e^{-D3/2})$$

$$\bullet \text{ DLH}_{LFO} = (\text{DLH}_{SS}/3100)^{0.6} (268 + 29K3 + 10.7K1 + 27K2)$$

$$\bullet \text{ DLH}_{DA} = 0.375(\text{DLH}_{LF/OPS}) (1 - 0.82 e^{-D4/3})$$

$$\bullet \text{ DLH}_{SAE} = 0.015 \times 10^{-3} (\text{DLH}_{SS})^{1.38}$$

$$\bullet \text{ DLH}_T + \text{DLH}_{QAR} = 425 e^{1.25 \times 10^{-4} \text{DLH}_{SS}^4}$$

PIONEER-CLASS JUPITER MISSION COSTS

A Cost Model Demonstration Exercise

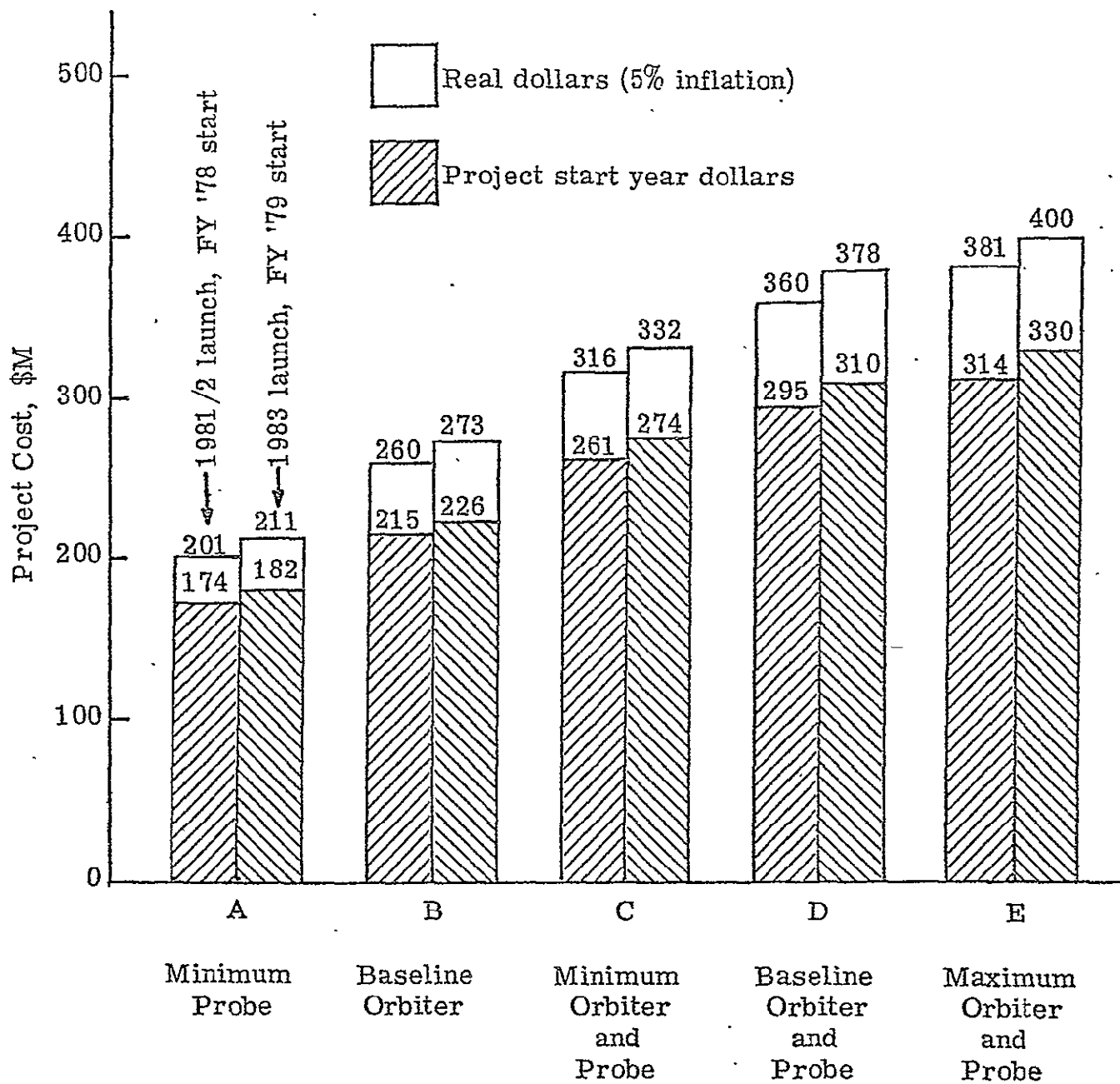
• GUIDELINES

- Pioneer-class spacecraft (spin-stabilized)
- 1981/2 and 1983 launch opportunities
- Orbiter and atmospheric probe missions
- Systems contractors (spacecraft and probe) project mode
- Conservative cost estimation (no hardware inheritance or standardization)
- Project start year dollars (1978 or 1979)

• MISSION OPTIONS

- A: Jupiter Probe (minimum concept)
- B: Jupiter Orbiter (baseline concept)
- C: Jupiter Orbiter/Probe (minimum concepts)
- D: Jupiter Orbiter/Probe (baseline concepts)
- E: Jupiter Orbiter/Probe (maximum* orbiter, baseline probe)

*includes imaging and planetology science



Mission Options

FIG. 7

COST COMPARISON SUMMARY OF PIONEER-CLASS JUPITER MISSIONS

TABLE 2

Option D
JUPITER ORBITER/PROBE (baseline concepts)

Mission Highlights

Baseline F/P bus science; baseline probe science
Flower orbit tour; day-side entry
72-month mission; 36 months in orbit
Shuttle/IUS (3) launch

Mass Summary

Science	51.1 kg	} 29.9 orbiter 21.2 probe
Spacecraft	349.5	
Probe	128.8	
Total Dry Mass	<u>529.4</u>	
Propellant ($\Delta V = 2270$ m/sec)	474.0	
Net Injected Mass	<u>1003.4 kg</u>	

Cost Estimate Summary

Science (instruments and data analysis)	\$ 36.5M
Spacecraft	110.2
Probe	48.3
Mission Operations	53.6
MCCC	13.6
Program Management and Design Analysis	13.4
Contingency	<u>19.2</u>
Total (FY '78 dollars)	\$294.8M
(FY '79 dollars)	309.5
(FY '75 dollars)	\$237.3M

2.3 Planetary Missions Performance Handbook--Vol. II, Inner Planets (1458 Man-Hours)

The purpose of the Planetary Missions Performance (PMP) Handbook series is to provide planetary program planners with the basic performance data which is essential in the preliminary stages of mission selection and planning. In the past, two types of NASA handbooks have been prepared for mission analysis work: 1) raw trajectory data handbooks such as the NASA SP-35 series, and 2) propulsion system performance handbooks such as the NASA Launch Vehicle Estimating Factors Document. The PMP Handbook series carries performance analysis one step further by combining these two basic groups of data in a form which is directly applicable to mission planning. The basic presentation format for one-way transfer data for inner planet missions is net payload versus launch window extent. Subsidiary pages show parametrized studies of performance sensitivity to change in orbit size and navigation impulse budget. The Mars Surface Sample Return section departs from this format to present a set of missions with parameters chosen according to a specified scheme. This scheme was established to pick the minimum mission configuration required for a particular launch opportunity, mission mode, and launch vehicle. This choice is minimum in the sense that it includes the least number of a set of fall-back steps, each of which is designed to produce mass relief for the mission. Results are presented in terms of mass margins available for Earth launch, Mars landing, and the Earth Return Vehicle (ERV).

Throughout all sections, the Handbook has been organized and assembled in such a manner as to permit revisions and additions to mission definitions and assumed propulsion capabilities, thus assuring continued usefulness and application to mission planning problems.

Volume II of the PMP Handbook series contains payload performance of missions to the inner planets. The scope of missions presently covered is shown in Table 3. Launch opportunities to Venus occur approximately every 1.6 years (19.2 months) and to Mars every 2.14 years (25.6 months). Trajectory characteristics for Venus opportunities display a cyclic behavior because the relative Earth-Venus transfer geometry repeats almost

TABLE 3

DATA SCOPE OF PMP HANDBOOK, VOL. II

MISSIONS	LAUNCH OPPORTUNITIES					
VENUS FLYBYS (TYPE I/II)	1981	1983	1984/5	1986	1988	1989
VENUS ORBITERS (TYPE I/II)	1981	1983	1984/5	1986	1988	1989
MARS FLYBYS (TYPE I/II)	1981/2	1983/4		1986	1988	
MARS ORBITERS (TYPE I/II)	1981/2	1983/4		1986	1988	
MARS SURFACE SAMPLE RETURN	1981/2	1983/4		1986	1988	1990

exactly every 8 Earth years. Thus, performance data for the 1981 and 1989 Venus missions will be very nearly equal in all cases. A similar situation obtains for Mars, but the orbital resonance is 15/7 (15 Earth years), and the match is not nearly so close as it is for Venus.

For the one-way transfers, Type I and Type II trajectories, which are characterized by heliocentric central angle less than or greater than 180 degrees, respectively, are distinguished and performance summaries for them are presented separately.

The propulsion systems which have been selected to define payload performance fall into two classes: 1) launch vehicles, and 2) retro stage systems. Table 4 shows all of the selected launch vehicles and the two rubber retro stages which are to be used for orbit capture at target. The period of application is shown for each of these, and presumes availability of all three Shuttle/IUS candidates by 1980-81, and both Tug configurations by 1985. The Atlas/Centaur is not shown past 1986 since current plans suggest that the Space Tug will be used for all planetary exploration missions as soon as it is available for this purpose.

Figure 8 outlines the organization of each of the one-way transfer sections. There are four of these, separated by tabs and paginated independently of each other for referencing convenience. For each, an introductory subsection briefly describes the trajectory characteristics of the mission and presents a summary of performance for a representative launch vehicle, the Shuttle/IUS(II). Then, for each opportunity, performance is presented as a function of launch window extent for each of the launch vehicles considered. Figure 9 is a sample of this format. Here, for the 1981 Mars opportunity, candidate orbiter missions are shown for a fixed orbit size and retro stage. The number on each curve references a particular launch vehicle, whose name may be found in a fold-out glossary at the rear of the Handbook. For example, #9 represents the Shuttle/IUS (II). Thus, Figure 9 shows that the Shuttle/IUS(II) applied to this mission would require a launch window of 15 days or less to place a 700 kg payload in the specified orbit.

TABLE 4

PROPULSION SYSTEMS FOR PMP HANDBOOK, VOL. II

ITEM	PERIOD OF APPLICATION									
	1981.	82	83	84	85	86	87	88	89	90
1) LAUNCH VEHICLES										
Atlas/Centaur	X									X
Atlas/Centaur/TE364-4	X									X
Shuttle/IUS (I) (12,915)*	X									X
Shuttle/IUS (II) (15,485)*	X									X
Shuttle/IUS (III) (25,575)*	X									X
Shuttle/Tug (R)/EE-Kick (29,500)*						X				X
Shuttle/Tug (E) (27,980)*						X				X
2) (RUBBER) RETRO STAGES										
Earth Storable	X									X
Space Storable	X									X

*Total cargo bay mass (kg); excludes payload, but includes 35 kg for adapter; recoverable Tug candidate must be off-loaded, such that cargo mass with payload = 29,500 kg.

I. INTRODUCTORY MATERIAL

- A. COMPARISON OF TRAJECTORY CHARACTERISTICS
FOR ALL LAUNCH OPPORTUNITIES (TABULAR)
- B. LAUNCH YEAR EFFECT ON PERFORMANCE (GRAPHICAL)

THEN, FOR EACH OPPORTUNITY---

II. GENERAL CHARACTERISTICS FOR THE OPPORTUNITY

- A. PERTINENT TRAJECTORY DATA AT FIXED
LAUNCH DATE INCREMENT FOR IMMEDIATE
REGION OF INTEREST (TABULAR)

III. PRACTICAL PERFORMANCE

- A. NET PAYLOAD VS. LAUNCH WINDOW EXTENT (GRAPHICAL)
- B. PERFORMANCE TABLES FOR ALL APPLICABLE
LAUNCH VEHICLES (TABULAR)

IV. IDEAL PERFORMANCE (ORBITERS ONLY)

- A. NET PAYLOAD VS. LAUNCH DATE FOR O-DAY
WINDOW EXTENT (GRAPHICAL)

FIG. 8

PMP HANDBOOK, VOL. II: DETAILED SECTION CONTENTS

1981/2

TYPE II TRANSFER

1981/2

EXCESS DV: 100 M/SEC
 ORBIT SIZE: ALT. = 1000 KM; CIRCULAR
 9.8 ORBITS/DAY

RETRO SYSTEM: EARTH-STORABLE: ISP = 300 SEC

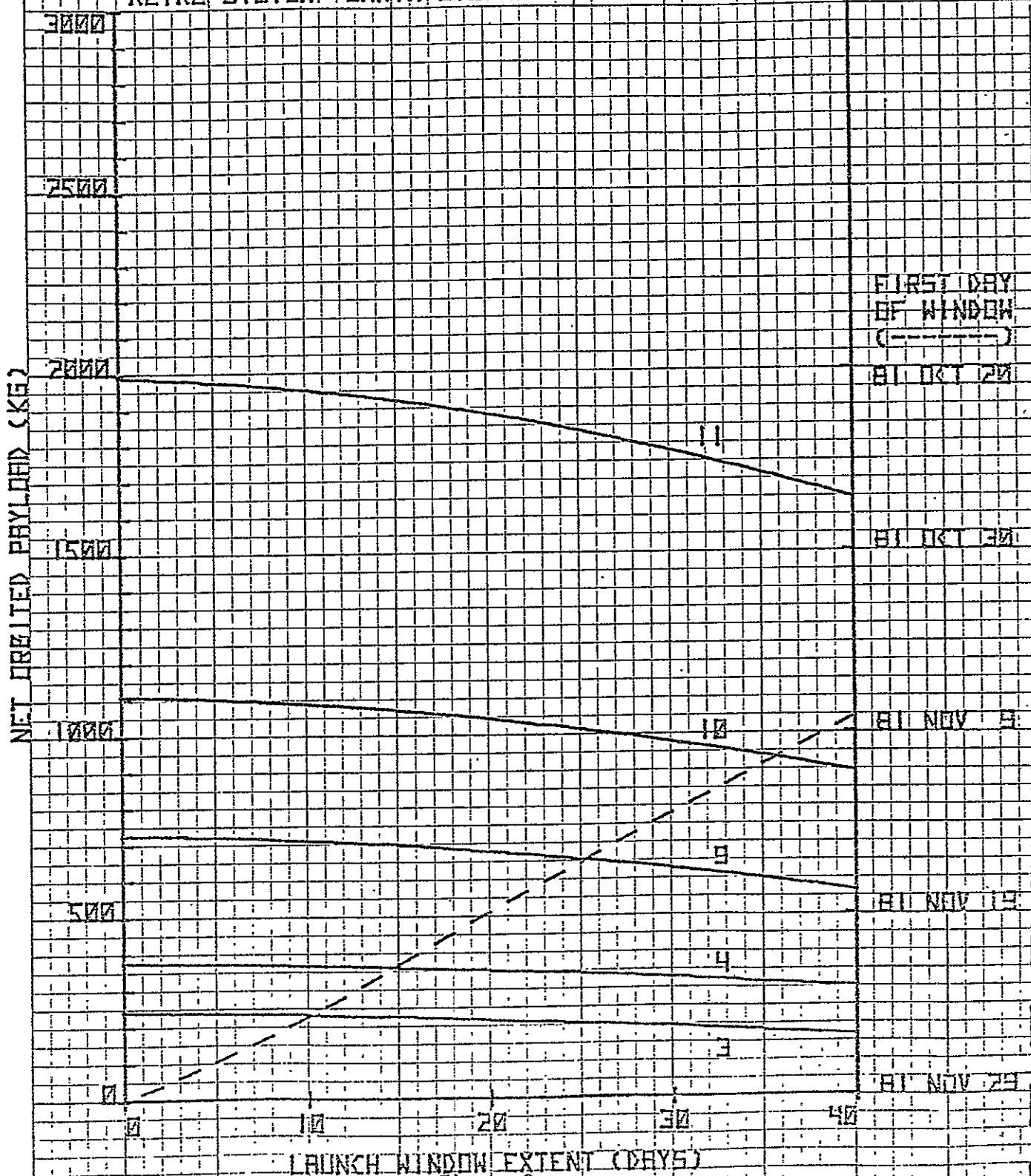


Figure 9

1981/2 MARS ORBITER
 MISSION PERFORMANCE SUMMARY

These graphs also show, as a dashed line, first day of the launch window as a function of window extent. A 15-day window for the example above would begin on 22 November 1981. For the flyby missions, the only additional data shown is trajectory data which characterize the near-optimum transfer region. For orbiter missions, tables are shown which present extensive trade-offs of injected payload vs. window extent and orbit size. For each choice of these parameters, the table shows net useful payload, and the retro stage mass required to place that payload in orbit. Thus, the sum of these two equals injected mass capability of the launch vehicle for the launch window extent. Finally, each orbiter section shows ideal (zero-day window) performance as a function of launch date in the region of near optimum performance. The contrast between this ideal and practical performance is explained below.

Initial transfer analysis was performed using the MULIMP trajectory optimization program. These results were then expanded into opportunity windows. Performance criteria chosen to optimize trajectories are minimum C3 for Flyby cases and minimum ΣV_{∞} for Orbiter cases. Then, for some particular window extent, the largest values of C3, declination of launch asymptote (DLA), and hyperbolic velocity at the target (VHP) are chosen to specify the window. The performance analysis is then based upon this somewhat conservative window definition. It turns out that, while the approach is conservative, it is nevertheless a good approximation to a window chosen with respect to a fixed retro stage size. The ideal performance curves are included to show upper bounds on performance throughout the region in which the launch windows are located.

The launch vehicle capability assumes due easterly launches from ETR. However, the analysis includes reductions in injected payload for non-easterly launches. For the expendable Atlas-based vehicles, a non-easterly launch penalty is imposed if DLA is greater than 28.5 degrees. An additional dog-leg maneuver penalty is also imposed if DLA is greater than 52.4 degrees. For Shuttle-based vehicles, only a dog-leg penalty is taken for DLA greater than 43.5 degrees, and then only if the Shuttle cargo mass exceeds capability at the associated launch azimuth.

A brief summary of results from all four sections is shown in Figures 10-13. For all of these, the launch window is fixed at 20 days and the Shuttle/IUS(II) is chosen as the launch vehicle. Orbit sizes are noted on the figures as applicable, and the performance increase of a space-storable retrosystem is indicated above the corresponding Earth-storable result.

The Mars Sample Return section presents mass performance summaries for the five opportunities in the period 1981-1990. Performance of three candidate launch vehicles, subject to their availability in this period (see Table 4) is displayed in tabular format in terms of available mass margin at certain critical points of the mission sequence. These tables also include descriptive information about the mission, including an impulse summary and flight time data. The tabular format is discussed in detail below.

For each launch opportunity, four basic mission modes are examined. They distinguish the four permutations upon Mars entry mode and Mars departure mode (see Table 5 below). Note that the so-called "direct" Mars departure is actually a departure from the extended stopover (~400 days) parking orbit. It is direct as opposed to the alternative orbital rendezvous of an orbiting bus with a planetary excursion module.

TABLE 5
MISSION OPTIONS

Option	Mars Entry	Mars Departure
1) D/D	Direct	Direct*
2) D/MOR	Direct	Orbital Rendezvous
3) O/D	Via Orbit	Direct*
4) O/MOR	Via Orbit	Orbital Rendezvous

* - from parking orbit

LAUNCH VEHICLE: SHUTTLE/IOU (11)
LAUNCH WINDOW: 20 DAYS

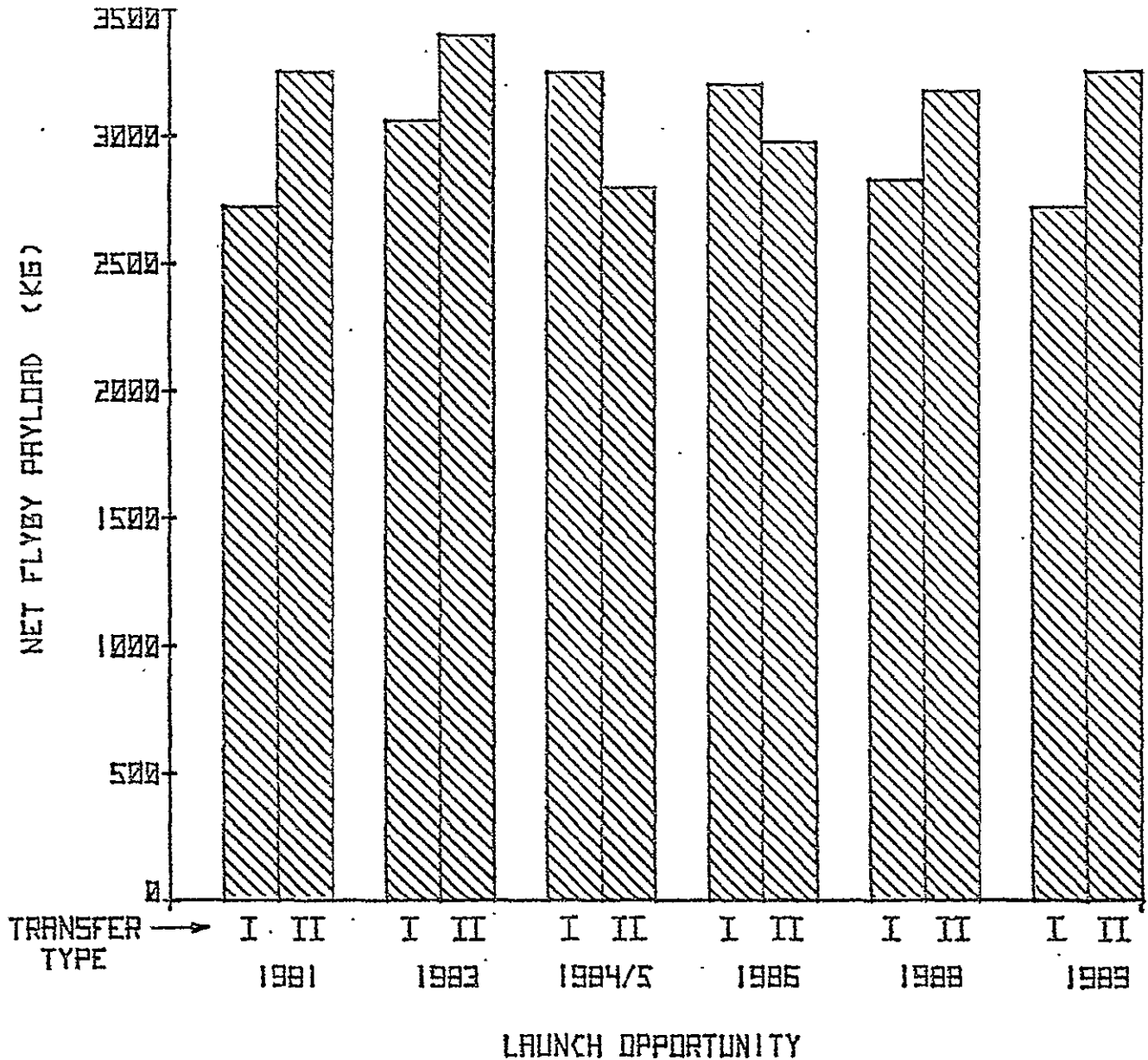


FIG. 10 VENUS FLYBY MISSIONS
LAUNCH YEAR EFFECT ON PERFORMANCE

LAUNCH VEHICLE: SHUTTLE/IUS (11)
 LAUNCH WINDOW: 20 DAYS
 ORBIT SIZE: CIRCULAR; ALT. = 500 KM

RETRO SYSTEM: EARTH-STORABLE - 
 SPACE-STORABLE - 

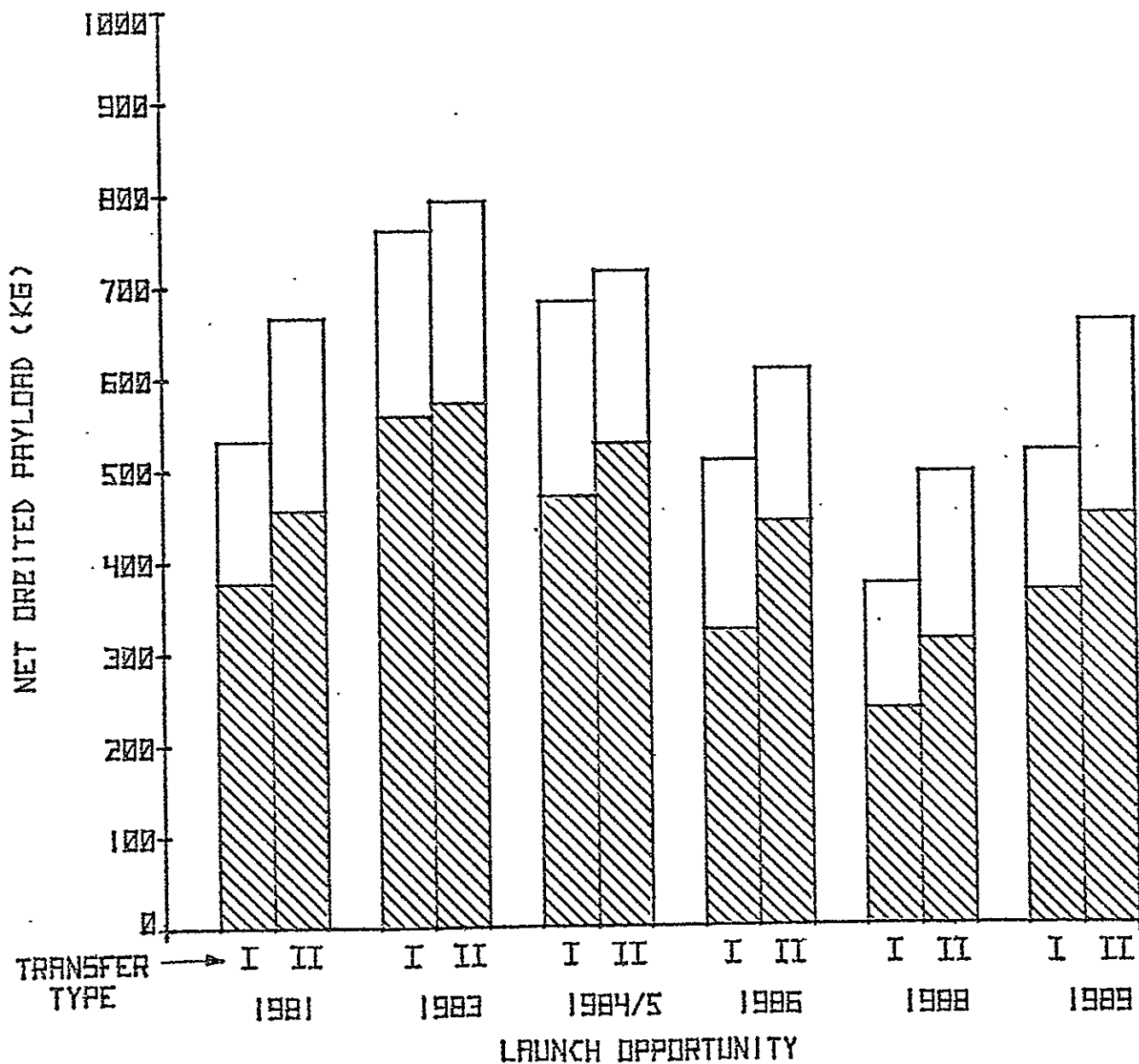


FIG. 11 VENUS ORBITER MISSIONS
 LAUNCH YEAR EFFECT ON PERFORMANCE

LAUNCH VEHICLE: SHUTTLE/105 (11)
LAUNCH WINDOW: 20 DAYS

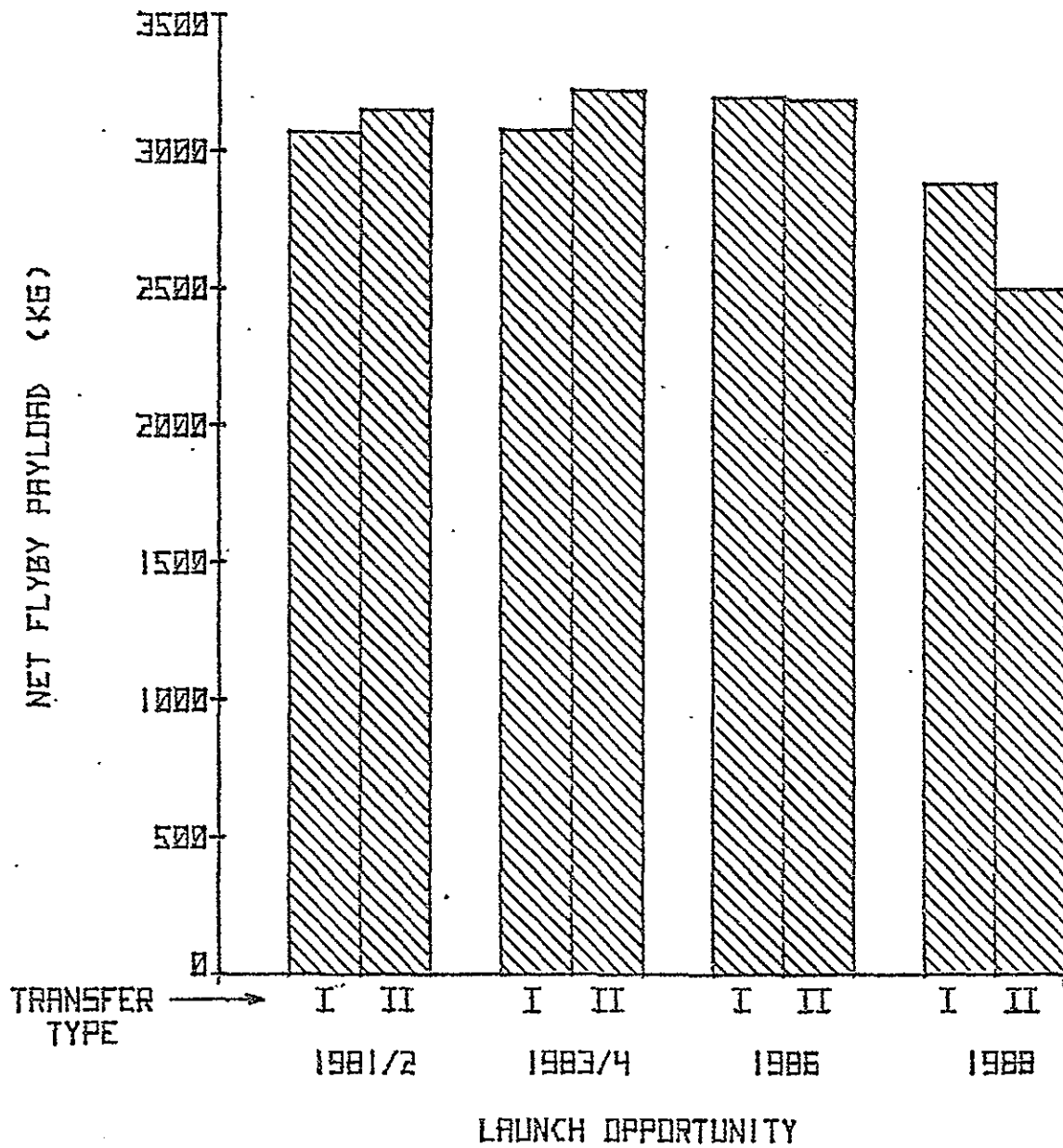


FIG. 12 MARS FLYBY MISSIONS
LAUNCH YEAR EFFECT ON PERFORMANCE

LAUNCH VEHICLE: SHUTTLE/105 (11)
 LAUNCH WINDOW: 20 DAYS
 ORBIT SIZE: CIRCULAR; ALT. = 1000 KM

RETRO SYSTEM: EARTH-STORABLE - 
 SPACE-STORABLE - 

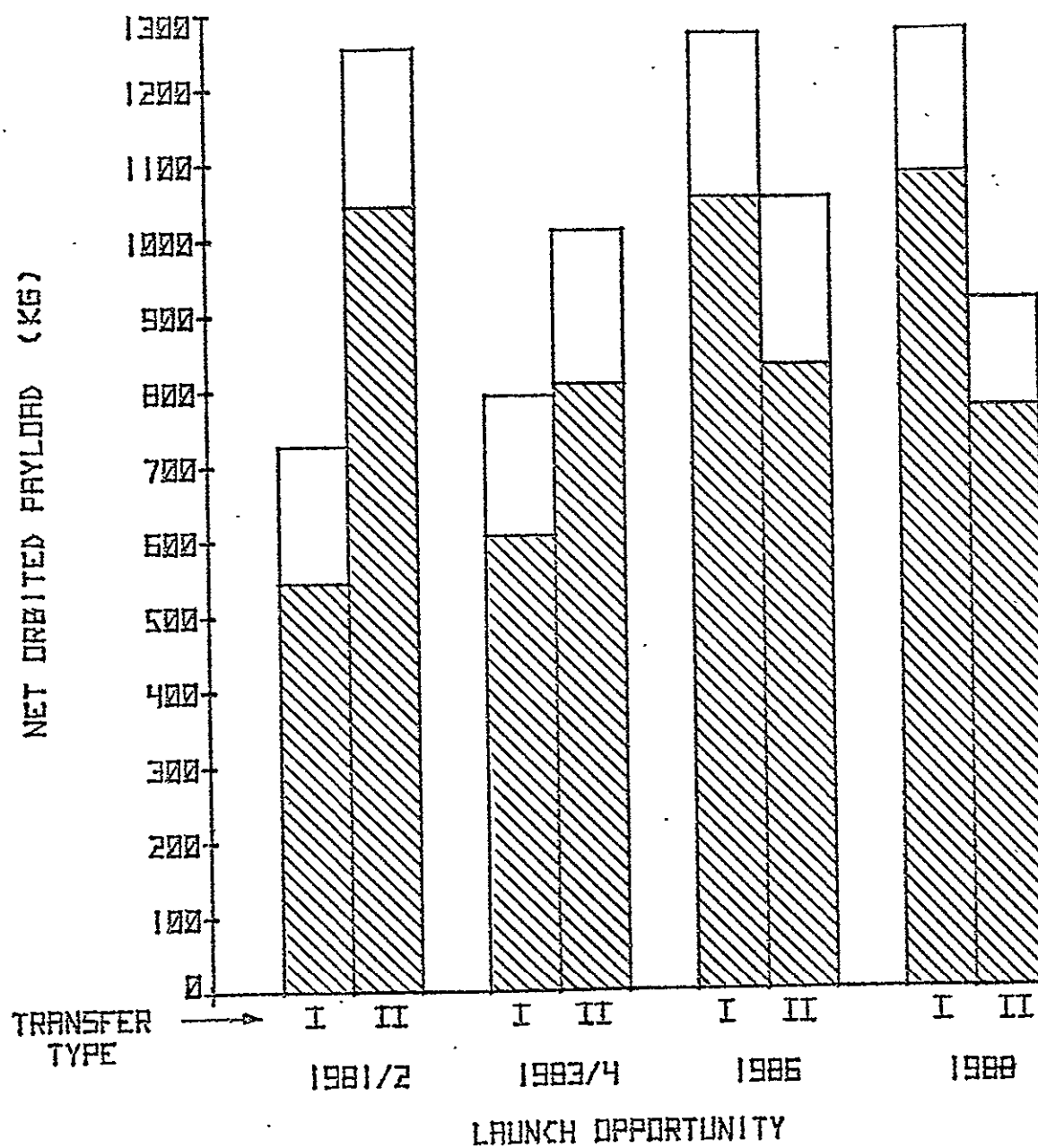


FIG. 13 MARS ORBITER MISSIONS
 LAUNCH YEAR EFFECT ON PERFORMANCE

The four mission modes are further defined by the scaling assumptions presented in Table 6. These assumptions and other data (orbit sizes, ΔV budget) are taken from recent studies of the sample return problem.

From these data, an analysis sequence was performed to produce any available mass margin at launch. The various velocity impulses and mass changes are "backed up" from Earth arrival through Mars departure, Mars arrival, and finally to Earth departure. Thus, the trans-Mars injection mass required to perform the mission is calculated and compared against the payload available with the selected launch vehicle. Any existing launch margin can then be propagated forward and applied to either the landed weight on Mars or the Earth return vehicle. In all cases, the entire margin is applied at the point selected.

For the initial search for a launch mass margin, it is assumed that Earth entry is via orbit capture, and that the Earth launch window is zero days in extent. Also, as applicable, it is assumed that the Mars retro system is Earth-storable and that the landing orbit is the one-day orbit. If these assumptions do not produce a margin at launch, the analysis is repeated with a prescribed order of fallback options applied until a margin is produced or the fallback procedure is exhausted. Each fallback step is designed to produce mass relief. In their order of application, they are:

- 1) 5-day landing orbit at Mars,
- 2) Space-storable Mars retro system, and
- 3) Direct Earth entry.

The 5-day landing orbit applies only to the two orbit entry modes, and the space-storable retro to all but the Direct entry/Direct return mission.

The fallback procedure is halted as soon as a "reasonable" Earth launch margin appears, where reasonable is taken to mean approximately 300 kg. Subsequent reversal of the analysis sequence applies the

TABLE 6

SCALING ASSUMPTIONS

- Sample size - 1 kg.
- Transfers - conjunction class
- Fixed masses
 - Earth entry capsule $\left\{ \begin{array}{l} \text{direct entry} - 30 \text{ kg.} \\ \text{orbit capture} - 20 \text{ kg. dry mass} \end{array} \right.$
 - ERV dry mass - 87 kg.
 - dry cruise bus (Mars direct entry only) - 230 kg.
 - dry orbiter bus (no MOR) - 550 kg.
 - dry orb iter bus (MOR) - 735 kg.
- Mass Scaling Relationships
 - 1) total landed mass at Mars $\cong 71\%$ of Mars entry mass
 - 2) Mars ascent vehicle (MAV) mass
 - $\cong 72\%$ of landed mass
 - $\cong 51\%$ of Mars entry mass
 - 3) MAV payload = 10% of MAV mass
- Retro Propulsion Systems
 - Mars orbit insertion - Earth storable (with space storable as an option)
 - Mars ascent - 2-stage solid
 - Earth orbit insertion - solid
- Mars entry altitude - 244 km.
- Earth entry altitude - 122 km.

margin (all of it) to the landed mass at Mars, and then to the Earth Return Vehicle. Table 7 shows a sample output format for the 1984 orbit entry/MOR case, using the 3-stage Shuttle/IUS configuration. The first block shows launch margin availability. In this instance, the baseline configuration produces a negative margin, so a 5-day landing orbit is used which improves the margin figure, but not enough. The next line shows application of a space-storable retro system for Mars orbit capture. Here, an acceptable margin has been found, so the window extent is expanded to 10 and then 20 days. At 20 day extent, the launch margin is still acceptable, so this case is chosen for presentation.

The mission description summarizes events, times, and the various velocity requirements. (Direct entry cases show entry velocity, rather than ΔV .) The last block shows the available launch mass margin applied at three different points in the mission sequence, and how masses at various critical points must be changed accordingly. In this case, application of the margin to the ERV makes no difference in the entry mass because all ERV mass (except the sample and its cannister) is assumed to remain with the orbiter bus. Alternatively, if the margin is applied to the landed mass, then no change is required in the orbiter.

As can be seen from the above example, the launch margin may be applied to either the ERV, or to the lander, but not both. This is the case for both MOR mission modes, but not for the Direct returns. The Direct return cases require that all ERV mass be landed on the planet, so application of the full margin to the ERV will also require all available landed mass margin.

Tables 8 through 11 show summaries by launch opportunity of launch vehicle masses and the various margins for each of the four mission modes. The launch vehicle is the Shuttle/IUS(III) which is available for all five Earth launch opportunities. The baseline configuration is used in all cases, save those where notes indicate otherwise. Comparison of these tables shows that the Direct Entry/MOR mode is almost always the most capable. Further, this mission mode can be flown in every opportunity with

no need of any of the fallback steps. In contrast, the Orbit/Direct mode is uniformly the poorest choice. In fact, the expendable TUG configuration is required to fly this mode, and even then, it can be done only in the last three launch opportunities considered.

It is anticipated that future additions to Volume II of the PMP Handbook series will include missions to Mercury, Venus Sample Return, and consideration of other sample return mission modes. A task just started will update Volume I (Outer Planets) with shorter launch windows, recently defined IUS propulsion selections, and new mission concepts such as VEGA/ Δ VEGA outer planet missions and satellite-assisted capture for Jupiter orbiters. New Jupiter swingby missions to Uranus, Neptune, and Pluto will also be added.

TABLE 7

1984

1984

MARS SURFACE SAMPLE RETURN

MASS PERFORMANCE SUMMARY

MISSION OPTION ORBIT ENTRY/MOR

LAUNCH VEHICLE SHUTTLE/IUS (III)

SAMPLE SIZE 1 KG

MASS MARGIN AVAILABLE AT EARTH LAUNCH

EARTH ENTRY OPTION	MARS RETRO SYSTEM	MARS ENTRY ORBIT	WINDOW EXTENT	LAUNCH MASS (KG)		
				REQ.	AVAIL.	MARGIN
ORBIT	EARTH-STORABLE	1-DAY	0 DAYS	5417	5265	-152
ORBIT	EARTH-STORABLE	5-DAY	0 DAYS	5275	5265	-9
ORBIT	SPACE-STORABLE	5-DAY	0 DAYS	4514	5265	749
ORBIT	SPACE-STORABLE	5-DAY	10 DAYS	4605	5208	603
ORBIT	SPACE-STORABLE	5-DAY	20 DAYS	4735	5140	405 ***

*** - THIS CASE IS DETAILED BELOW

MISSION DESCRIPTION

EVENT	OPTION	DATE	TYPE	MANEUVER	FLIGHT TIME
EARTH LAUNCH		5 JAN 1984	II	C3 = 11.882	EARTH-MARS LEG 281 DAYS
MARS ARRIVAL	ORBIT	12 OCT 1984		DV = 1.513	MARS STOPOVER 507
MARS DEPARTURE	MOR	3 MAR 1986	I	DV = 0.732	MARS-EARTH LEG 213
EARTH ARRIVAL	ORBIT	2 OCT 1986		DV = 2.337	
					TOTAL MISSION 1001 DAYS 2.7 YRS

CUMULATIVE MASS SUMMARY (KG)

MASS MARGIN APPLIED TO

	EARTH LAUNCH	MARS LANDER	ERV
EARTH ENTRY.....	44	44	44
EARTH RETURN VEHICLE	256	256	414 ←
MARS ASCENT.....	493	493	493
MARS LANDER.....	768	940 ←	768
MARS ENTRY.....	1205	1428	1205
EARTH LAUNCH.....	5140 ←	5140	5140
AVAILABLE MARGIN....	405 ←	172 ←	158 ←

TABLE 8

MARS SURFACE SAMPLE RETURN
LAUNCH OPPORTUNITY EFFECT ON PERFORMANCE

MISSION OPTION: DIRECT ENTRY/DIRECT RETURN

Launch Vehicle: Shuttle/IUS (III)

Mission Parameters*

Launch Windows: 20 days
Earth Entry: Orbit capture

		<u>OPPORTUNITY</u>				
		<u>1981/2</u>	<u>1983/4</u>	<u>1986</u>	<u>1988</u>	<u>1990</u>
Launch Vehicle Mass (kg)	{ Available	5846	5584	5525	5112	4852
	{ Required	4980	4538	5036	4766	4401
Mass Margins (kg)	{ Launch	507	1046	490	346	451
	{ Landed mass	376	777	364	253	335
	{ ERV	30	62	29	20	27
Notes					b	b

* - except as indicated by notes below

a - 10 day launch window

b - direct entry at Earth

TABLE 9

MARS SURFACE SAMPLE RETURN
LAUNCH OPPORTUNITY EFFECT ON PERFORMANCE

MISSION OPTION: DIRECT ENTRY/MOR

Launch Vehicle: Shuttle/IUS (III)

Mission Parameters*

Launch Window: 20 days
Earth Entry: Orbit capture
Mars Retro Stage: Earth-Storable

OPPORTUNITY

		<u>1981/2</u>	<u>1983/4</u>	<u>1986</u>	<u>1988</u>	<u>1990</u>
Launch Vehicle Mass (kg)	{ Available	5339	5140	5519	5018	4635
	{ Required	3861	4322	4641	3632	3675
Mass Margins (kg)	{ Launch	1478	818	878	1387	960
	{ Landed mass	1203	646	696	1124	762
	{ ERV	571	263	265	602	395
Notes						

* - except as indicated by notes below

a - 10 day launch window

b - direct entry at Earth

c - Space-Storable Mars Retro System

TABLE 10

MARS SURFACE SAMPLE RETURN
LAUNCH OPPORTUNITY EFFECT ON PERFORMANCE

MISSION OPTION: ORBIT ENTRY/DIRECT RETURN

Launch Vehicle: Shuttle/IUS (III)

Mission Parameters*

Launch Window: 20 days
Earth Entry: Orbit capture
Mars Retro Stage: Earth-Storable
Landing Orbit: 1 day period

OPPORTUNITY

		<u>1981/2</u>	<u>1983/4</u>	<u>1986</u>	<u>1988</u>	<u>1990</u>
Launch Vehicle Mass (kg)	{ Available	5455	5265	5625	5154	4780
	{ Required	5845	6828	8518	6868	6496
Mass Margins (kg)	{ Launch	-389	-1563	-2892	-1714	-1716
	{ Landed mass	-	-	-	-	-
	{ ERV	-	-	-	-	-
Notes		a,b,c,d	a,b,c,d	a,b,c,d	a,b,c,d	a,b,c,d

- * - except as indicated by notes below
- a - 0 day launch window
 - b - direct entry at Earth
 - c - Space-Storable Mars Retro System
 - d - 5 day Mars landing orbit

TABLE 11

MARS SURFACE SAMPLE RETURN
LAUNCH OPPORTUNITY EFFECT ON PERFORMANCE

MISSION OPTION: ORBIT ENTRY/MOR

Launch Vehicle: Shuttle/IUS(III)

Mission Parameters*

Launch Window: 20 days
Earth Entry: Orbit capture
Mars Retro Stage: Earth-Storable
Landing Orbit: 1 day period

OPPORTUNITY

		<u>1981/2</u>	<u>1983/4</u>	<u>1986</u>	<u>1988</u>	<u>1990</u>
Launch Vehicle Mass (kg)	{ Available	5339	5140	5519	5018	4635
	{ Required	4851	4735	5075	<u>4382</u>	4530
Mass Margins (kg)	{ Launch	489	405	444	636	105
	{ Landed mass	206	172	179	304	47
	{ ERV	188	158	164	275	43
Notes			c,d	c,d		

* - except as indicated by notes below

a - 10 day launch window

b - direct entry at Earth

c - Space-Storable Mars Retro System

d - 5 day Mars landing orbit

2.4 Penetrator Mission Concepts for Mercury and the Galilean Satellites (2000 Man-Hours)

Penetrators are elongated missile-shaped objects weighing about 35 kg and designed to implant scientific and electronic instrumentation to depths of 1 to 15 meters in a wide variety of soils. Surface impact is at a nominal speed of 150 m/sec oriented as close as possible to the vertical direction. The penetrator concept offers an exciting potential for conducting subsurface science investigations at various planetary bodies within the solar system. Potential data return includes soil dynamic properties and stratigraphy, heat flow, elemental chemistry, volatile analysis, and seismic activity. In comparison with large and complex soft landers such as Viking, penetrator missions should be intrinsically less costly--particularly for multiple site deployments.

The present analysis expands the horizons of earlier Mars application studies by focusing on Mercury and the four Galilean satellites of Jupiter. As these target bodies have no atmosphere to aid penetrator descent to the surface, more stringent requirements are imposed on the penetrator's deployment (retro) and control systems. Two additional factors which raise the level of mission difficulty relative to Mars are the longer trip times and more massive propulsion systems needed to deliver a spacecraft bus and penetrators to these targets. While it is important to underscore these points at the outset, it does appear that missions to Mercury and the satellites may be technically possible in the post-1985 time period. The conditional basis for this general conclusion will be summarized here and developed in detail in the main body of the report.

From the viewpoint of mission planning, the question of feasibility may be addressed on two levels--primary and secondary considerations.

Primary Feasibility: Is it reasonable to expect that the necessary operational functions leading to penetrator emplacement can be accomplished by appropriate design? In particular, can the penetrators be guided to required impact conditions with sufficient accuracy, and is the total system mass requirement within the capability of programmed launch vehicles?

Secondary Feasibility: Are the constraints imposed by practical mission design factors compatible with the relevance and performance of desired science experiments? In particular, can the shock and thermal environment be accommodated, are the accessible impact sites suitable, and is the data transmission capacity sufficient? Also, is the estimated cost of the mission reasonable in light of programmatic constraints and priorities of other mission types?

There is no intent to imply that the "secondary" considerations are of lesser importance, only that they would be of academic interest if primary feasibility cannot be established. This report attempts to answer the first-order questions and to provide information which hopefully will be useful to any follow-on assessment of the second-order questions.

The scope of this study encompasses mission concepts and alternative options for design implementation. Data are presented to characterize performance requirements and capabilities with a view towards assessing basic mission feasibility. Key areas of mission analysis are covered ranging from launch opportunities to penetrator data communications. To a large extent the analysis describes parametric design tradeoffs, although specific baseline selections are made when appropriate. An example of this is the mission profile description given in Table 12 which serves to integrate the various study elements. Supporting data relating to some of the more pertinent design parameters are summarized in Table 13.

Conceptually the mission profile is similar to that of a planetary orbiter up to the point of penetrator deployment and subsequent data collection. There are three major hardware elements comprising the total system: 1) the penetrators including their individual solid propellant retro motor; 2) a spacecraft bus or orbiter; and 3) a propulsion system which implements all post-launch trajectory maneuvers including orbit insertion at Mercury or Jupiter as the case may be. The bus provides all the usual mission operational functions (power, attitude control, communications, etc.) and may also include an orbiter science payload not necessarily related to penetrator objectives. However, the main function of the bus is to support the penetrator mission by: 1) controlling orbital maneuvers required for pre-deployment penetrator targeting;

TABLE 12

PENETRATOR MISSION BASELINE SELECTIONS

	<u>Mercury</u>	<u>Galilean Satellites</u>
Launch Opportunity	March 1988	June 1987
Launch Vehicle	Shuttle/Tug (Expendable)	Shuttle/Tug (Expendable)
Delivery Mode	Ballistic (Venus Swingby) Flight Time = 2 years	Ballistic (Direct Transfer) Flight Time = 2.6 years
Number of Penetrators	3	3
Spacecraft Bus (TBD)	Spinner 3-Axis	Spinner 3-Axis
Spacecraft Propulsion	E/S + Solid S/S	E/S E/S
Orbit Selection	P = 21 hours, h_p = 500 km I = 90°, North Polar Latitudes	Near-Equatorial Jupiter Orbit Ganymede-Assisted Capture Initial Period $\approx$ 100 days Orbit Period Pump Down
Penetrator Deployment	Out-of-Orbit, h_p Lowering Maneuvers Apoapse Release/Bus Retargeting	Swingby Encounter (Orbit Resonance) Rectilinear Approach/Bus Retargeting
Penetrator Retro	Solid, High Thrust, 4.1 km/sec	Solid, High Thrust, 3.5 km/sec
Penetrator Control	Attitude (closed-loop) Flight Path (open-loop)	Attitude (closed-loop) Flight Path (open-loop)
Impact Sites	Polar Latitudes Temperature Constraints	TBD (Science-Relevance) Orbit Resonance Constraints
Data Transmission	Potentially Very High	High/Low (Alternate Encounters) Limitation on Experiment bits/day Depending on Target Satellite

TABLE 13

PENETRATOR MISSION DESIGN PARAMETERS

	<u>Mercury</u>	<u>Ganymede^a</u>
Spacecraft ΔV (km/sec)		
Midcourse	0.279	0.050
Orbit Insertion	3.757	1.115
Orbit Maneuvers	0.250	0.450
Penetrator Retro		
ΔV (km/sec)	4.1	3.5
Thrust (newtons)	30,000	30,000
Ignition Altitude (km)	3	34
Burn Time (sec)	18.5	12.5
Penetrator Free-Fall		
Rest Altitude (km)	3.0	7.9
Fall Time (sec)	40	105
Angle-of-Attack Adjustment		
Max. Pitch-Over (deg)	90	23
Control Torque (newton-m)	0.36	0.36
Response Time (sec)	15	9
Penetrator Impact		
Speed (m/sec)	150	150
Max. Deceleration (g's)	1800	1800
Impact Errors (3σ)		
Speed (m/sec)	41	15
Path Angle (deg)	36	24
Angle-of-Attack (deg)	3.8	7.5
Miss Distance (km)	21	62
Data Transmission ^b		
RF Power (watts)	2	5
Input Energy (watt-hr)	0.75	6.2
Communications Time (min)	4.5	15/15
Data Transmitted (bits)	3.2×10^6	$2.8 \times 10^6 / 0.3 \times 10^6$
Average Accumulation (bits/day)	3.7×10^6	0.2×10^6

a. Other satellite data in report

b. Optimum communications geometry, single penetrator

2) providing a stable platform for penetrator release, one whose position relative to the target is well-determined; and 3) serving as a communications relay between the implanted penetrators and the Earth tracking stations. One additional function which may be highly desirable if prior information is lacking is impact region mapping (resolution of terrain uncertainty) leading to site selection. This implies an imaging system aboard the orbiter, the necessity of which is totally compatible with navigation accuracy requirements for these missions.

The penetrator system, on its part, is hardly immune from major functional requirements. Following separation from the bus, control responsibility is shifted entirely to the penetrator and continues to surface impact. Although the characteristics of baseline deployment modes at Mercury and the satellites are quite different, the general sequence of events is as follows: 1) attitude control prior to retro ignition; 2) mark retro ignition; 3) thrust vector control during retro-fire and measure thrust acceleration; 4) retro burnout and jettison at design rest altitude; 5) reorientation of longitudinal axis to align with predicted value of impact velocity direction; and 6) free fall to surface under active attitude control. A three-axis control scheme is necessary since spin stabilization is not suitable for elongated configurations which are dynamically unstable when rotating about their roll axis.

In principle the penetrator system can be made as complex as desired, even approaching a full-fledged lander. However, any significant move in this direction must be avoided if the mission concept is to remain at all viable. The design groundrule should be simplicity matched to adequate performance. The proposed control system hardware consists of lightweight inertial gyros and accelerometers in strapdown configuration, computational logic integrated with the central microprocessor controlling all penetrator operations, cold gas jet actuators for command and limit-cycle attitude control prior to and after the retro maneuver, and TVC actuation via a flexible bearing nozzle integrated with the solid retro design. An additional element that is required, particularly for satellite missions, is a lightweight radar altimeter used simply to mark

the retro ignition altitude.

Flight path control is "open-loop" in the sense that orbit determination (O.D.) information relating to the (fixed) direction of retro velocity is supplied by the orbiter bus at the time of penetrator release. Hence, after retro burnout there is no control of impact speed, path angle and miss distance; the dispersion in these parameters is primarily a function of O.D. and retro execution errors (1.5%, 1°). The most crucial impact parameter, angle-of-attack, is adjusted after retro burnout based on accelerometer measurements of the retro ΔV vector. This effectively compensates for the large residual velocity uncertainty due to execution error leaving only the smaller contribution of the O.D. and inertial sensor error sources. Angle-of-attack is controllable in this way to a maximum (3σ) error of 4° at Mercury and not more than 8.5° at the satellites; the nominal design constraint ranges from 3° to 11° depending on surface material hardness. One area of concern is the flight path angle error whose 3σ value exceeds the nominal constraint of 15° (seismic experiment requirement). It is noted however that from a statistical viewpoint the probability of satisfying this constraint is above 80%. Greater accuracy in executing the retro maneuver would eliminate the potential problem of experiment performance degradation.

Mass performance summaries for Mercury and satellite penetrator missions are presented in Tables 14 and 15. A ballistic flight mode is assumed in each case, and comparative results show the effect of an axis-stabilized versus a spin-stabilized bus design. These are to be considered generic spacecraft types typifying existing or planned designs; e.g., MVM vs PVO at Mercury and MJ0 vs PJ0 at Jupiter. The mass of the basic bus includes a typical orbiter science payload. The additional allowance of 50 kg for structural modifications to carry the penetrators is a zero-order estimate not verified by detailed analysis.

Mercury is the more difficult of the two missions. Shuttle/Tug launch vehicle capability is required for ballistic transfers by way of one or more gravity-assist swingbys of Venus. The alternative to the

TABLE 14

MERCURY PENETRATOR MISSION MASS SUMMARY*

• PENETRATOR SYSTEM

Basic Penetrator	35 kg
Guidance & Control	10
Retro Propulsion	<u>200</u>
Total Deployment Mass	245

• SPACECRAFT BUS SYSTEM (3 Penetrators)

Bus Retro Class	Axis-Stabilized		Spin-Stabilized	
	<u>E/S + Solid</u>	<u>S/S</u>	<u>E/S + Solid</u>	<u>S/S</u>
Basic Bus	460 kg	460 kg	270 kg	270 kg
Deployment Structure/Mechanisms	50	50	50	50
Orbit Maneuver Propellant	891	683	770	589
Orbit Insertion Propellant	5487	4250	4740	3672
Propulsion System Inerts	668	864	587	757
	<u>7556</u>	<u>6307</u>	<u>6417</u>	<u>5338</u>
Penetrators (3) + 10% Contingency	<u>810</u>	<u>810</u>	<u>810</u>	<u>810</u>
Total Launch Mass	<u>8366</u>	<u>7117</u>	<u>7227</u>	<u>6148</u>
Shuttle/Tug Performance	7550	7550	7550	7550
Shuttle/IUS Performance	4000	4000	4000	4000
Total Launch Mass				
2 Penetrators	6745	5740	5605	4770
1 Penetrator	<u>5130</u>	<u>4360</u>	<u>3990</u>	<u>3390</u>

*1988 Launch Opportunity, Ballistic Venus (2) Swingby

TABLE 15

GALILEAN SATELLITE PENETRATOR MISSION MASS SUMMARY*

• PENETRATOR SYSTEM

Basic Penetrator	35 kg
Guidance & Control	10
Retro Propulsion	<u>140</u>
Total Deployment Mass	185

• SPACECRAFT BUS SYSTEM (3 Penetrators)

Bus Launch Year	Axis-Stabilized		Spin-Stabilized	
	<u>1987</u>	<u>1990</u>	<u>1987</u>	<u>1990</u>
Basic Bus	700 kg	700 kg	700 kg	700 kg
Deployment Structure/Mechanisms	50	50	50	50
Orbit Maneuver Propellant	309	324	234	245
Orbit Insertion Propellant	853	1209	645	915
Propulsion System Inerts	226	274	189	225
Subtotal	<u>2138</u>	<u>2557</u>	<u>1468</u>	<u>1785</u>
Penetrators (3) + 10% Contingency	<u>612</u>	<u>612</u>	<u>612</u>	<u>612</u>
Total Launch Mass	<u>2750</u>	<u>3169</u>	<u>2080</u>	<u>2397</u>
Shuttle/Tug Performance	3630	2830	3630	2830
Shuttle/IUS Performance	1430	1070	1430	1070

ΔV-EGA FLIGHT MODE

Total Launch Mass	3785	2860
Shuttle/IUS Performance	3900	3900

*Earth-Storable Spacecraft Propulsion

conventional ballistic delivery mode is high-powered solar electric propulsion or other advanced low-thrust systems such as nuclear-electric or solar sails. For the ballistic mission baseline, it is found that space-storable retro propulsion for orbit insertion is needed to provide sufficient payload performance for 3 penetrator deployments off an axis-stabilized spacecraft bus. The most severe design constraint is the hostile thermal environment at Mercury which must be accommodated by both the orbiter bus and the implanted penetrator. Long lifetime penetrators powered by RTG's will have to be restricted to near-polar impact sites. Such sites are also most compatible with de-orbit maneuvers which utilize solar perturbation control, and high data transmission levels throughout the mission. For an impact site at 80°N latitude, the effect of Mercury's rotation causes the data transmission capacity to degrade by no more than a factor of 3 (-5dB) relative to the ideal communications geometry. A 2 watt transmitter could support an average daily accumulation of science data between 1.2×10^6 and 3.7×10^6 bits throughout a long lifetime mission. Short lifetime, battery-powered penetrators may be feasible in the equatorial region near the warm poles; more detailed analysis relating to battery, electronics and instrumentation response would be necessary to confirm this.

Galilean satellite missions could be planned for most any launch year opportunity to Jupiter utilizing direct ballistic transfers. Shuttle/Tug capability is again required. In the event that only the Shuttle/IUS were available, it would then be necessary to resort to one of the indirect flight modes (ΔV -EGA, VEGA or SEEGA) at the expense of much longer trip times to Jupiter. Earth-storable retro propulsion appears to be adequate for purposes of Jupiter orbit insertion, but space-storables do provide a greater margin of payload performance and launch opportunity flexibility. The thermal control problem for satellite missions is that of maintaining sufficiently high operating temperatures in the relatively cold ($100\text{--}150^{\circ}\text{K}$) ambient environment. The amount of heat that would be added by the RTG's is very uncertain at present because of inadequate knowledge and possibly large variations in the subsurface thermal conductivity. Insulation

material and additional heat sources may be required for relatively high conductivity, icy regions of impact. On the other hand, impact into silicate material of low conductivity may require a design approach currently favored for a Mars penetrator, i.e. a variable conductivity heat pipe dissipator. The limitations on an accurate thermal analysis would be alleviated by improved knowledge of satellite thermophysical properties. Anticipated remote sensing experiments on earlier satellite flyby missions would be helpful in this regard.

Another potential problem area for satellite penetrators is the inherent limitation on data transmission capacity for RAM deployment, i.e. when the spacecraft bus is in orbit about Jupiter and reencounters the target satellite at relatively infrequent intervals. Assuming a single penetrator and a 5-watt transmitter, the average accumulation of science data that could be supported ranges from 80,000 bits/day at Callisto to 450,000 bits/day at Io. This capability is further reduced when one considers multiple emplacements at any given satellite. It is somewhat doubtful that this data level could support a viable seismic net experiment, even with sophisticated scheme of data discrimination and compression. Possible improvements in the communication link are offered by: 1) variable transmission rate which optimally tracks the signal strength received by the bus; 2) increasing the transmitter power; and 3) replacing the assumed isotropic bus antenna by a high gain directed antenna which tracks the penetrator location. These areas are recommended for further study. It should be noted that the above-mentioned limitations on data communications are removed if the bus could be placed into orbit about the target satellite. This option has been investigated and found to be only marginally possible from a mass performance standpoint.

Preliminary cost estimates for each mission have been made for two different scenarios relating to spacecraft bus development. The first assumes a totally new development, the second assumes maximum inheritance derived from a "block buy and modify" option which might be possible if there were a precursor orbiter mission programmed a few years before the penetrator mission. The two estimates could be viewed as upper and lower

bounds on expected cost. Cost element breakdowns are given in the report; Table 16 summarizes the results for total mission cost (excluding launch vehicle) given in FY '77 dollars. For the Mercury mission the cost bounds are \$211-317 million assuming an axis-stabilized bus, or \$167-232 million for a spin-stabilized bus. Cost estimates for the Galilean satellite mission are slightly higher by about 10%.

An overall conclusion based on the results of this analysis is that penetrator deployment at Mercury and the Galilean satellites warrant further consideration as possible mission options for the late 1980's. This presumes the availability of Shuttle/Tug launch vehicle capability or its equivalent, and also the expected technology of fairly large and high performance space propulsion systems. Design problems relating to deployment and subsurface operation do not seem insurmountable in light of the high-level of spacecraft design experience and new technology trends in retro propulsion, navigation, and electronics. It would be prudent however to place these mission concepts in perspective with the proposed Mars penetrator: a) fewer penetrators per spacecraft; b) more limited accessibility of impact sites; c) possibly lower daily accumulation of science data; d) greater risk; and e) higher cost.

One of the more important areas lacking precise definition is science experiment requirements for Mercury and satellite targets. Science considerations relate both directly and indirectly to many aspects of the baseline mission definition. Some clarification in this regard is expected from the recently formed ad hoc Surface Penetrator Science Committee. Another area of uncertainty is the generic type of spacecraft bus most suitable for carrying penetrators and supporting mission operations. Spacecraft designs for future orbiters of Mercury and Jupiter are still in the evolution phase; it would be timely to factor in the special requirements of penetrator deployment. Finally, it is noted that the possible advent of advanced propulsion delivery systems such as solar electric propulsion could replace the more conventional ballistic flight mode baselined here for Mercury. This would obviously impact other

TABLE 16

COST ESTIMATE FOR PENETRATOR MISSIONS

- 1 Flight Article
- FY '77 Dollars
- LV Costs Excluded

	<u>Mercury</u>	<u>Galilean Satellites</u>
<u>Axis-Stabilized Bus</u>		
New Development	\$316.8 M	\$340.2 M
Maximum Inheritance	\$211.2 M	\$226.5 M
 <u>Spin-Stabilized Bus</u>		
New Development	\$232.1 M	\$256.6 M
Maximum Inheritance	\$167.1 M	\$182.6 M

baseline design choices from launch opportunity through penetrator deployment; the most important gain would be a vast reduction in the size of the propulsion system needed for Mercury orbit insertion.

2.5 Geology Orbiter Comparison Study (890 Man-Hours)

In the cost-conscious mid 1970's, the concept of a single basic spacecraft design equipped with a standard instrument payload for the study of the surfaces and interiors of all the bodies in the solar system has a certain appeal. Many spacecraft systems such as those for communications exhibit commonality and can be used in a variety of missions with little adaptation. But is the same true of scientific instrument packages? Is it possible to define a set of instruments with wide application in geological exploration to the geological exploration of the planets? Could the Lunar Polar Orbiter payload, for example, be adapted with very little modification for remote sensing missions to the terrestrial planets and Jovian satellites? If such an adaptation is possible, how do these instruments perform under these varying circumstances? How does the spacecraft design and space environment affect performance? Can new instrument designs and mission concepts be identified that minimize the negative aspects and exploit the opportunities presented by missions outside the Earth-Moon system? These are the primary questions we attempt to answer in this geology orbiter comparison study.

Two basic discipline areas in the general field of the crustal and interior properties of the solar system bodies are considered (see Table 17). In geochemistry, the discipline area is surface elemental composition and three remote sensing techniques are considered. They are gamma ray, X-ray fluorescence and atomic spectroscopy. In geophysics, the field is planetary and satellite interiors and the technique considered is the complementary radar/gravity experiment.

In the field of geochemistry and for the inner planets Mars, Moon and Mercury, gamma ray spectroscopy is a powerful technique with the ability to provide quantitative measurements of many geochemically significant elements. X-ray spectroscopy at the Moon and Mercury is limited to the light elements up to silicon because of the abrupt energy cut-off in the stimulating solar X-ray flux, and is not effective at Mars because of atmospheric attenuation. X-ray spectroscopy can provide higher spatial

TABLE 17
GEOLOGY ORBITER COMPARISON STUDY

<u>EXPERIMENT SET EVALUATED</u>		
<u>DISCIPLINE</u>	<u>PRIME SCIENTIFIC OBJECTIVES CONSIDERED IN STUDY</u>	<u>TECHNIQUES CONSIDERED IN STUDY</u>
GEOCHEMISTRY	DETERMINATION OF ELEMENTAL ABUNDANCES IN SURFACES OF TERRESTRIAL PLANETS AND GALILEAN SATELLITES	GAMMA RAY SPECTROSCOPY X-RAY FLUORESCENCE SPECTROSCOPY ATOMIC SPECTROSCOPY IN THE ULTRAVIOLET VISUAL AND NEAR INFRARED
GEOPHYSICS	DETERMINATION OF TOPO- GRAPHIC AND GRAVITATIONAL PROPERTIES OF TERRESTRIAL PLANETS AND SATELLITES AS A MEANS OF DEDUCING INTERNAL MASS DISTRIBUTION AND ISOSTATIC STATE	RADAR/DOPPLER TRACKING RADAR/GRAVITY GRADIOMETER

resolution information than X-ray methods given the same amount of observing time. X-ray spectrometers suitable for the Lunar Polar Orbiter are easily adapted for missions to Mercury and performance is largely controlled by the orbital constraints. Gamma spectrometers suitable for lunar observations require little modification for operation at Mars but significant redesign in order to accomplish passive detector cooling in the severe thermal environment of Mercury orbit. Performance characteristics for X-ray and gamma-ray remote sensing have been generated for circular orbiters of the Moon and Mars and an elliptical Mercury orbiter.

Observations of gamma and fluorescent X-rays originating from the Galilean satellites can also lead to the detection of geochemically significant elements. The most important source of excitation of these emissions is trapped magnetospheric radiation. Long range or observatory mode observations from outside the most intense regions of the Jovian radiation belts will be more effective than close encounters of the Galilean satellites with the exception of observations of Callisto. Resonant emission at visible wavelengths of atomic species derived from the surface of the satellite Io has already been observed from Earth and can also be conducted in observatory mode with a spectrometer of sub angstrom spectral resolution. Elemental sensitivity is enhanced by an order of magnitude over that achievable from Earth orbit, and three dimensional mapping can be conducted with three orders of magnitude better spectral resolution than is achievable from Earth or Earth orbit.

Quantitative estimates of surface elemental abundances from gamma, X-ray and visible line emission are highly model dependent. Thus, these remote sensing techniques must be examined very critically as far as the realization of any specific geochemical objectives is concerned. On the other hand, these remote sensing techniques can allow the interaction of the satellites with magnetospheric particles to be probed in great detail. Observations conducted from Earth and from the Pioneer flybys has revealed a rich interplay of physical phenomena to be taking place between satellites and magnetosphere. The relevance of these phenomena to

comparative planetology is yet to be demonstrated. Nevertheless, they are clearly accessible to study with remote sensing and a Jupiter orbiter equipped with remote sensing instrumentation could carry out the observations. Lunar Polar Orbiter instrument designs are not directly applicable to these types of observation.

In our analysis of geophysical methods we find that a possible Lunar Polar Orbiter derivative beam-width-limited radar can be applied successfully to the investigation of all the planets except Venus. Definition of the shape and topography is constrained by mission design in the cases of Mercury and the Galilean satellites. State-of-the art gradiometers offer no advantages in the reconnaissance of the satellites but Doppler tracking will yield definitive measurements of J_2 and J_3 for reasonable flyby velocities, and possibly also other gravitational harmonics if mission design flexibility permits.

2.6 Performance Handbook for Shuttle-Based Lunar Missions (406 Man-Hours)

The purpose of this task is to prepare a handbook for rapid payload performance assessment of Shuttle upper stages applied to lunar missions of planning interest for the 1980's. This study grew out of the need to determine the impact on future lunar missions of the wide range of upper stages proposed for the Shuttle. Even with the current commitment to the solid IUS design, specific stage parameters have not yet been fixed and other stages (e.g., the Tug) are still under consideration for future application. Hence, this handbook is expected to be a valuable aid in determining near-term and future lunar mission capabilities.

The scope of the Handbook covers four types of missions: orbiters, halo orbiters, landers, and sample returns. Two types of Shuttle upper-stages are considered: expendable and recoverable. Both single and double expendable stages are included, while single recoverable stages were assumed. Combining the mission and stage types led to a total of seven cases to be considered for the performance analysis. Each of these cases is defined in Table 18. The characteristics of Cases 1-5 are explained by the data provided in Table 18. Case 6, Sample Retrieval, and Case 7, Sample Collection/Return, differ in that Case 6 assumes that a sample return payload is waiting in lunar orbit for the arriving recoverable stage. Case 7, on the other hand, requires that the arriving recoverable stage bring with it the necessary hardware to collect a sample from the lunar surface and deliver this back to it in orbit. The main difference between these two cases then is the larger payload and longer staytimes required by the latter.

The method of approach used to determine the required performance data had as a guideline the desire to keep the number of independent parameters to a minimum, thus holding the volume and complexity of the final results at a workable level. To this end each of the mission cases (and several subcases) defined above were analyzed for their characteristic velocity requirements. From these results the cases were grouped

TABLE 18

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CASE DEFINITIONS

Case	Translunar Flight Time (hrs)	Lunar Stay Time (hrs)	Expended Stage	Recovered Stage	Stage Return Time (hrs)
1 Halo Orbiters	128-140	-	X		
2 Halo Orbiters	128-140	-		X	36-72
3 Orbiters/Landers	96	-	X		
4 Orbiters/Landers	96	-		X	36-72
5 Orbiters/Landers	72*	-		X	144
6 Sample Retrievals	66	12		X	144
7 Sample Coll./Returns	60	24		X	144

*free-return stage recovery

into three performance classes: low, medium, and high--subsequently referred to as Classes 1, 2, and 3, respectively. A summary of the individual case requirements and resulting class velocity requirements is presented in Table 19. In the process of reducing the number of mission options to three classes, three cases (2B, 4B, and 5) have been eliminated because they provided little or no performance advantage over other similar cases, and had some other less desirable characteristics, e.g. longer stage return time. A fourth case, Case 7, has been eliminated because its velocity requirements proved to be in excess of what even a stage with Tug-level technology could provide, operating within the constraint of a single Shuttle launch.

The formats selected for presentation of the performance data are of two types: one for single stage applications (both expendable and recoverable) and the other for two stage applications (expendable only). For single stage applications, the stage parameters of interest are initial mass, stage mass fraction, and vacuum specific impulse, in this order. Hence, performance graphs were prepared of net payload versus stage mass with curves of constant stage mass fraction. For any one plot the performance class (and hence characteristic mission velocity requirement) and specific impulse are held constant. An example of this type of plot is shown in Figure 14 for a Class 1 mission. Note that the retro propulsion parameters stated in the upper left hand corner, along with the constant stage parameter, are fixed. These parameters were held constant throughout the analysis, representative of current Earth-storable technology. Note also, as shown in the title of Figure 14, that this plot is for lower stage masses, i.e., Lo Mass Scale. A companion plot exists for each parameter variation for higher stage masses, i.e., Hi Mass Scale. To illustrate the use of these performance plots assume, for example, that the payload performance of the TE 364-4 Kick Stage is to be determined for a Class 1 Lunar mission. This stage has an ignition mass of 1260 kg and a stage mass fraction of 0.86. Assuming the stage can deliver an Isp of 290 sec (the Handbook contains data for four different Isp's), one quickly determines from Figure 14 that the

TABLE 19

Lunar Performance Handbook
CHARACTERISTIC VELOCITY REQUIREMENTS

Case	Stage Return Time (hrs)	Characteristic Δv 's(m/sec)		Performance Class	
		Stage(s)	Payload Retro	Level	Δv Budget (m/sec) ^a
1	--	3185	800 ^b	Lo	3200/1000
2A	36	6610	800 ^b	Med	6650/1000
2B	72	6645	800 ^b	---	eliminated ---
3	--	3200	975	Lo	3200/1000
4A	36	6630	975	Med	6650/1000
4B	72	6660	975	---	eliminated ---
5	144	6620	1010	---	eliminated ---
6	144	9430	--	Hi	9450/0
7	144	9530	2080/1870 ^c	---	eliminated ---

- a. Δv budget expressed as two numbers: stage/payload retro.
 b. Retro budget includes 3 years of halo orbit station-keeping.
 c. Landers require 2080 m/sec to reach the surface from lunar orbit;
 return to lunar orbit for sample pick-up requires an additional 1870 m/sec.

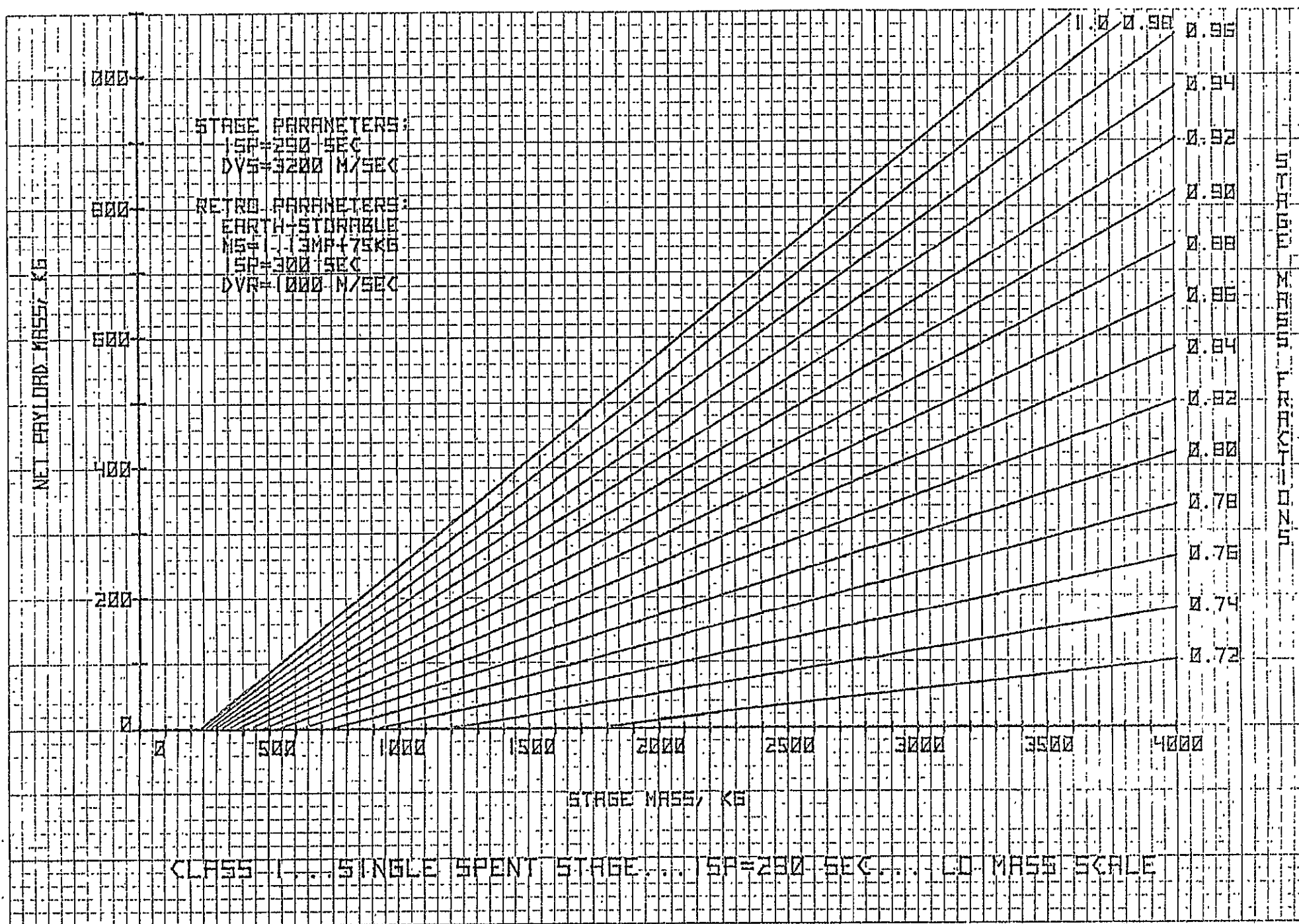


FIG. 14

payload capacity of this stage is 155 kg. Furthermore, it can be seen that if the stage inerts are reduced improving the mass fraction to 0.90, for example, the payload will increase to 205 kg. The sensitivity of payload to mass fraction for this case directly follows being 12.5 kg per point (0.01) of mass fraction.

The second format used to display payload performance is associated with two-stage applications which are restricted to Class 1 missions. An example of this type of graph is shown in Figure 15. Net payload is plotted against first stage mass for curves of constant second stage mass. More stage parameters must necessarily be held constant for this type of plot and include the Class impulse, stage Isp's and mass fractions, and retro propulsion assumptions. However, more graphs have been prepared to show the sensitivity of performance to Isp and variable mass fractions. As with the single stage data, both Lo and Hi Mass Scale graphs are presented for each set of assumed Isp's and mass fractions. Figure 15 is an example of the Hi Mass Scale data.

The full range of parameterized performance results contained in the Handbook is summarized in Table 20. Each bullet represents a performance graph; a total of 40 graphs have been prepared. An appendix of conversion data is also included in the Handbook to expand the usefulness of these data. The appendix data permits the user to redetermine payload performance using either a monopropellant or space-storable retro system in place of the baseline earth-storable retro system assumed. Another graph is included to convert orbited payload to landed payload for an assumed descent propulsion system. Finally, several graphs are included to permit the user to determine initial stage mass and stage mass fraction (the required stage input data for determining performance) from other stage definitions, e.g. propellant loading and dry weight.

The Handbook is organized to combine the advantage of easy usage with the helpfulness of a lunar missions propulsion requirements summary. Assumptions, scope, and method of approach are discussed in the Introduction. The scope of missions are then summarized and

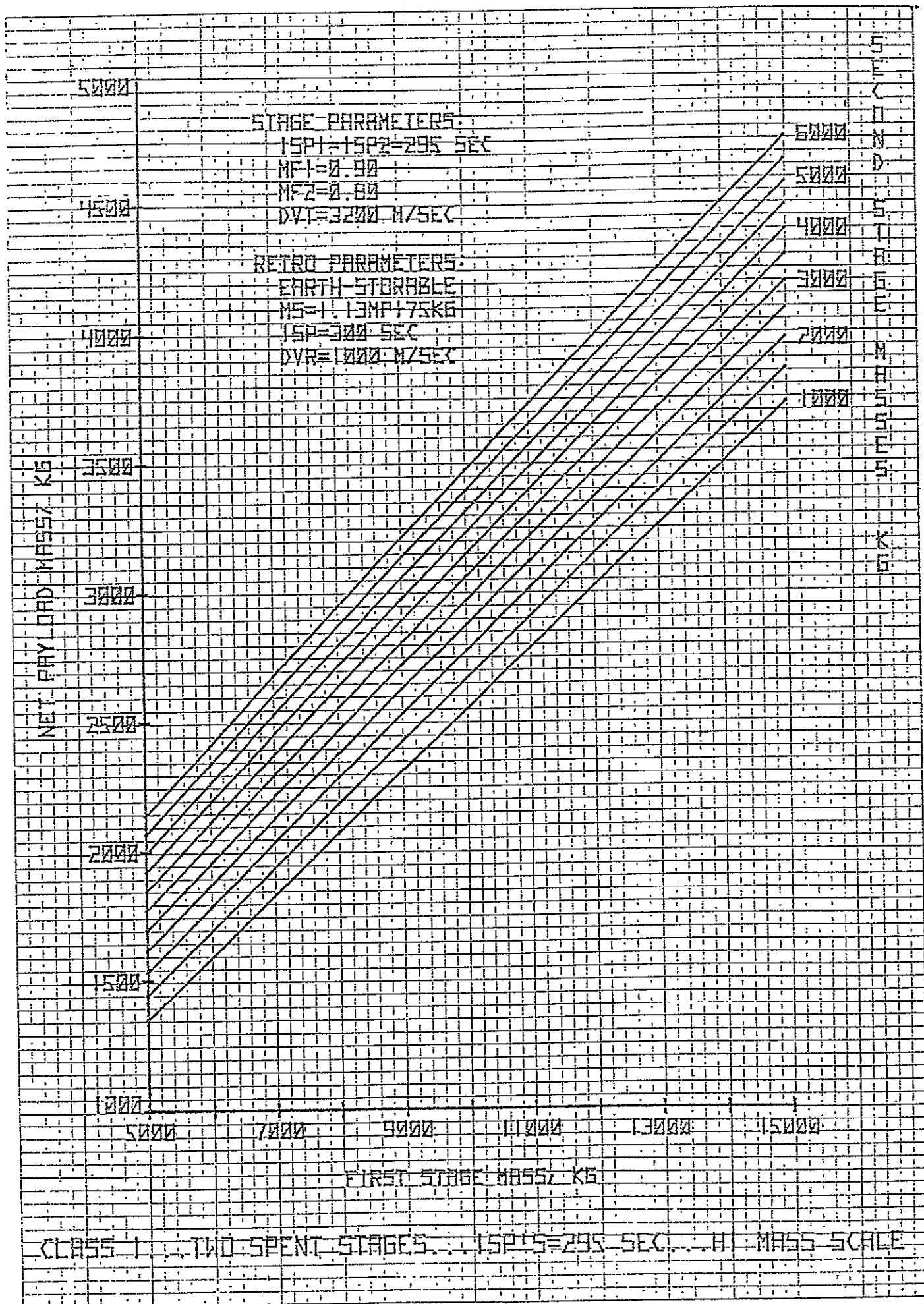


FIG. 15

TABLE 20

Lunar Performance Handbook

PERFORMANCE PARAMETER SCOPE

Performance Class	Stages	Mass Fractions	Mass Scales	Propulsion Specific Impulses, Isp, (sec)					
				285	290	295	300	445	460
Lo	1	0.72-1.0	low						
			high						
	2	0.70/0.85	low						
		0.75/0.85	low						
		0.80/0.85	low						
		0.85/0.55	low						
		0.90/0.70	high						
		0.90/0.75	high						
		0.90/0.80	high						
		0.90/0.85	high						
Med	1	0.72-1.0	low						
		0.72-1.0	high						
Hi	1	0.88-1.0	low						
	1	0.88-1.0	high						

classified. Finally, payload performance data is presented for each of the three Mission Classes with individual data indexes defining the extent of stage parameter variations covered within each class.

3. REPORTS AND PUBLICATIONS

Science Applications, Inc. is required, as part of its advanced studies contract with the Planetary Programs Division, to document the results of its analyses. This documentation traditionally has been in one of two forms. First, reports are prepared for each scheduled contract task. Second, publications are prepared by individual staff members on subjects within the contract tasks which are considered of general interest to the aerospace community. A bibliography of the reports and publications completed during the contract period 1 February 1975 through 31 January 1976 is presented below. Unless otherwise indicated, these documents are available to interested readers upon request.

3.1 Task Reports for NASA Contract NASW-2783

1. "Penetrator Mission Concepts for Mercury and the Galilean Satellites," Report No. SAI-1-120-399-M5; February 1976.
2. "Planetary Missions Performance Handbook--Volume II, Inner Planets," Report No. SAI-1-120-399-M6, February 1976.
3. "Manpower/Cost Estimation Model for Automated Planetary Programs--2," Report No. SAI-1-120-399-C2, April 1976.
4. "Comparison of Geology Orbiter Experiments for Planetary Exploration," Report No. SAI-1-120-399-S1, May 1976.
5. "Performance Handbook for Shuttle-Based Lunar Missions," Report No. SAI-1-120-399-M4, May 1976.
6. "Advanced Planning Activities, February 1975-January 1976," Report No. SAI-1-120-399-M7, April 1976.
7. "Advanced Planetary Studies Third Annual Report," Report No. SAI-1-120-399-A3, April 1976.

3.2 Related Publications

1. "New Directions in Automated Spacecraft Cost Estimation," P. P. Pekar, A. L. Friedlander and D. L. Roberts, Journal of Spacecraft and Rockets, 12, 8, pp. 458-464, August 1975.

2. "Pioneer Mars 1979 Mission Options," J. C. Niehoff and A. L. Friedlander, Synoptic in Journal of Spacecraft and Rockets, 12, 8, pp. 451-452, August 1975.
3. "Measurement Error Analysis in Determination of Small-Body Gravity Fields," A. L. Friedlander, D. R. Davis and T. A. Heppenheimer, Synoptic in Journal of Spacecraft and Rockets, 12, 6, pp. 325-326, June 1975.
4. "Multi-Asteroid Flyby Trajectories Using Venus-Earth Gravity Assists," D. F. Bender and A. L. Friedlander, AAS Paper No. AAS 75-086, AAS/AIAA Astrodynamics Specialist Conference, Nassau, Bahamas, July 28-30, 1975.
5. "Jupiter Orbiter Lifetime--The Hazard of Galilean Satellite Collision," A. L. Friedlander, AAS Paper No. AAS 75-038, AAS/AIAA Astrodynamics Specialist Conference, Nassau, Bahamas, July 28-30, 1975.